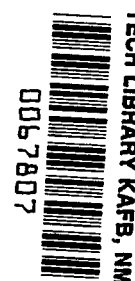


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Preliminary Tests of an Advanced High-Temperature Combustion System

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National Aeronautics
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Scientific and Technical
Information Branch

Summary

A combustion system has been developed to operate efficiently and with good durability at inlet pressures to 4.05 MPa (40 atm), inlet air temperatures to 900 K, and exhaust gas temperatures to 2480 K. The developed combustion system incorporates a dump diffuser, two annular, single-zone fuel injection rows with low-injection-pressure fuel modules, and a film-cooled, louvered combustor liner.

A preliminary investigation of this system was conducted at inlet pressures to 0.94 MPa (9 atm), an inlet air temperature of 560 K, and exhaust gas temperatures to 2135 K. A maximum combustion efficiency of 98.5 percent was attained at a fuel-air ratio of 0.033; the efficiency decreased to about 90 percent as the fuel-air ratio was increased to 0.055. This investigation is described in this report.

At an exhaust gas temperature of 2090 K the average of all combustor liner thermocouples indicated a metal temperature 355 kelvins greater than the nominal inlet air temperature. The maximum local liner temperature was about 565 kelvins above the nominal inlet air temperature and decreased to 505 kelvins above as combustor pressure was increased.

Tests to determine the isothermal total pressure loss of the combustor showed that at a diffuser inlet Mach number of 0.35, the system loss was 6.5 percent; 2.6 percent of that is attributable to the fuel modules and blocking plate, and 1.1 percent to the combustor liner.

Introduction

Compressor pressure ratios and turbine inlet temperatures of aircraft gas turbine engines have been increasing steadily since the introduction of these engines. The trend is toward high-pressure, high-temperature gas generators. Cycle and mission analyses have indicated that compression ratios as high as 40 with turbine inlet temperatures to 2480 K may be required for future advanced engines (ref. 1). The air exiting a 40:1 compression ratio compressor will be at about 900 K. This high-temperature air with its reduced heat sink capacity will have to cool metal engine parts that will be subjected to increasing heat fluxes because of higher turbine inlet temperatures. These metal parts, such as turbine blades and vanes and combustor liners, will thus require better cooling techniques and will also require alloys with better high-temperature durability characteristics.

Necessary design information regarding metal temperature levels and gradients, combustor exit temperature profiles, and other durability and performance characteristics associated with advanced engine cycles

cannot be obtained through extrapolation of data acquired in current engines or test rigs. A research facility was therefore constructed in order to obtain realistic experimental design criteria for the cooling of metal parts that will operate at advanced engine conditions. As part of the facility a combustion system was required to furnish the desired advanced engine environment to the turbine rig test equipment. It was designed for operation at 4.05-MPa (40-atm) pressure, with inlet air and exhaust gas temperatures to 900 and 2480 K, respectively.

The results of preliminary tests of this combustion system in the combustor rig of the same research facility are presented in this report. In these tests the combustion system was operated at inlet pressures to 0.94 MPa (9 atm), a nominal inlet air temperature of 560 K, and exhaust gas temperatures to 2135 K. At these conditions the combustor exhibited good performance and durability.

Apparatus

Research Facility

A research facility to simulate the high temperatures and high pressures of aircraft turbine engine inlet and exit conditions was constructed at the NASA Lewis Research Center. The facility consists of independent and parallel combustor and turbine blade and vane test rigs with supportive service systems. An integrated system of minicomputers and programmable controllers provides automated control, safety monitoring, and data acquisition for the entire facility. Details of the operation of the facility, data acquisition, and data recording are given in reference 2. The maximum operating condition that the facility can supply to the inlet of a test rig is 4.05-MPa (40-atm) pressure, 900 K temperature, and 82-kg/sec flow.

Figure 1, an overall perspective of the facility, shows the air preheater, the air compressors, and the two test rigs. The heated, pressurized air is routed to either test rig by a swing elbow that physically isolates the rig not under test. Also shown are the approximate positions of the venturi airflow-measuring station and the pressure and flow control valves.

Dimensional details of the piping that ducts the pressurized, heated air to the combustor rig are shown in figure 2, which also includes the positions of inlet temperature and pressure instrumentation planes. Airflow straightener plates are used in the inlet piping to evenly distribute the air entering the combustor.

Test Rig

The combustor test rig inlet and exit housings are shown in figures 3 and 4. The flow is from left to right. Figure 3 shows the inlet section, the inlet air temperature-

and pressure-measuring planes, the fuel injection plane, and the exhaust gas quench tank. Figure 4 shows the rotating instrumentation termination plate from which instrumentation leads, pressure tubes, and instrumentation rake cooling water are ducted into the center shaft of the quench tank. Additional cooling water, steam trace lines, and gas sample lines are ducted into the center rotating shaft through rotary seal connections. All rotating instrumentation, instrumentation rake cooling water, steam trace lines, gas sample lines, and main cooling-water lines are hard wired or hard tubed inside the quench tank.

A cross-sectional view of the combustor test rig is shown in figure 5. The positions of the inlet air and exhaust gas instrumentation and the fuel injection planes are indicated.

Research Hardware

Inlet diffuser. — The combustor inlet diffuser, shown in figure 6, is a full-annular dump diffuser with eight equally spaced struts. The diffuser is 7.62 cm long with an area ratio of 1.79. Inlet total and static pressures are sampled 5.23 cm upstream of the diffuser inlet.

Fuel modules. — The fuel-module designs used in these tests are shown in figure 7. Each fuel module consisted of two concentric air swirlers that swirled the air in opposite directions to create a zone of high shearing action (fig. 7(a)). All vanes were at an angle of 45° to the axial direction. Fuel was supplied to each module by a fuel tube located in the central cavity of each module (fig. 7(b)). Fuel flowed from the fuel tube through an 0.084-cm-diameter discharge opening and impinged on a splash plate mounted on the downstream face of each module. This splash plate broke up the fuel jet and directed it radially outward, where the fuel was further atomized by air passing through the inner swirler. Additional fuel atomization occurred in the shearing region between flows exiting the counterrotating air swirlers. The two rows of fuel modules with the inner combustor liner installed are shown in figure 8. Also shown are the outer-row module support struts and the blocking plate between modules.

Combustor liners. — The separate inner and outer liners are shown in figure 9. The two liners, when fitted together, formed an annular combustion zone, as shown in figure 6. Each liner was constructed of seven overlapping panels. Cooling air was directed onto the back side of the trailing edge of a panel by a row of film-cooling-air holes and was then discharged along the hot-gas side of the next downstream panel in the form of a cooling film. The combustor liner overall length was 23.9 cm, and the length from the fuel-module discharge plane to the exhaust instrumentation plane was 22.0 cm. The maximum cross-sectional area (reference area) of the combustor annulus was 0.2474 m².

Detailed thermal, stress, and film-cooling analyses were made with 1260 K as the maximum allowable liner metal temperature. Results indicate that Hastelloy X with a thickness of about 0.20 cm is a suitable liner material. The analyses were made for uncoated metal surfaces. Before the tests the liners were coated with a thermal-barrier coating. Reference 3 gives details of the coating and the method of application.

Combustor liners similar to the liners used in the present investigation were film airflow calibrated at ambient pressure and temperature to determine the validity of assumptions made for calculating the film-cooling-air flow rate. A sketch of the calibrating stand is shown in figure 6 of reference 4. Each row of holes could be calibrated separately, or the entire liner could be calibrated with all of the rows unblocked.

Table I lists the number of holes, the hole diameter, and the total flow area for both inner and outer liners. Using the flow coefficients listed in table I of reference 4 obtained during film-airflow calibration of liners of similar geometry permitted the calculation of film cooling air to the combustor liners used in the present investigation.

Research Instrumentation

Inlet instrumentation. — Airflow rates were measured by a venturi installed according to ASME specifications. Transducers with various differential pressure ranges were used to measure the venturi pressure differential. The position of the venturi in the combustion air system is shown in figure 2.

Fuel flow rates were measured by both turbine and strain gage flowmeters. The strain gage type uses the drag force of the flowing medium across a special target. The fuel was ASTM Jet-A.

Combustor inlet air instrumentation was mounted at planes 2 and 3, as shown in figures 2, 3, 5, and 6. Position details of the inlet air Chromel-Alumel thermocouples are shown in figure 10. The indicated readings of the inlet air thermocouples were taken as true values of the total temperature. The dimension details of the four total pressure rakes of three probes each, positioned at centers of equal areas, are shown in figure 11. Also shown are the positions of four static pressure probes, or taps, mounted on the outer wall of the combustor housing.

Two static pressure probes were located at both the upstream and downstream sides of the fuel-module blocking plate. One pair was installed at a circumferential position of 123° and the other pair at 243° . The approximate location at the blocking plate of one pair is shown in figure 6. Also shown in figure 6 are the approximate axial positions of the four liner annulus static pressure probes: two in the inner annulus and two in the outer annulus. All four were located at a circumferential position of 32° .

Combustor liner instrumentation. — Chromel-Alumel thermocouples were installed on both combustor liners to obtain liner average metal temperature. The thermocouples were installed so as to indicate the hot-gas-side surface temperature of the metal, the side on which the thermal barrier had been applied. There were 16 thermocouples on each liner, and the liner average metal temperature was determined from the arithmetic average of the 32 thermocouples.

Exhaust instrumentation. — Exhaust instrumentation was mounted at plane 4 (figs. 5 and 6). Two gas sample rakes, two total pressure rakes, and two temperature rakes were mounted on a rotating drum forming the inner wall of the exhaust annulus. Similar rakes were mounted 180° apart. Since the drum rotated 180° both clockwise and counterclockwise, the two rakes provided full 360° coverage of the exhaust annulus. The five sampling ports on each rake were canted across the annulus, instead of positioned radially, because of space limitations. Four static pressure probes were also located at the sampling plane. The rotating drum could be programmed to move in either 3°, 6°, or 9° steps or in a continuous sweep mode. At each step of the rotating drum, data could be sampled, displayed on local readout meters, and recorded. During sweep operation data could be monitored on real-time readout equipment. The drum could also be stopped at any desired position to determine variations in data with time or to investigate areas of abnormal instrument readings.

Gas sample rake: The five-port gas sample rakes were mounted as shown in figure 12. Ports 2 to 4 (inner diameter to outer diameter) were at centers of equal areas. Ports 1 and 5 positions had to be varied a small amount because of physical construction constraints caused by cooling requirements (fig. 12). The five sample ports were each connected to a common plenum by about 5 cm of 0.16-cm-outer-diameter stainless steel tubing. Cooling water flowed across the outside of these tubes. The plenum pressure was controlled to about 0.27 MPa by a venting system. The sample gas then entered a steam-traced stainless steel sample line (sample temperature controlled to about 425 K) that ducted the sample to the gas analysis consoles. Valving permitted analysis of the sample from either rake alone or from both rakes combined.

The gas analysis instruments were designed to operate in automatic, semiautomatic, and manual modes from a remote-control panel. The system could measure carbon monoxide, oxygen, water vapor, total hydrocarbons, carbon dioxide, and oxides of nitrogen. Microprocessors were used to set the ranges of the analyzers, to calibrate the system, to process the raw data to units of concentrations, and to make the data and other system information available to the operator through a terminal and control panels. The operation of the consoles is described in detail in reference 5.

Total pressure rake: The mounting positions of the two rakes are shown in figure 12. As with the gas sample rakes, ports 1 and 5 (inner diameter and outer diameter) were somewhat displaced from the centers of equal areas; ports 2 to 4 were at the proper positions. Each rake port was connected to both a transducer and a Scanivalve port.

Temperature rake: The mounting positions of the two temperature rakes are shown in figure 12. The five nonaspirating thermocouple junctions of each rake were at centers of equal areas. The thermocouples were constructed from 0.025-cm-diameter Pt/Pt-13%Rh wire and had a nongrounded junction. They were contained within a 0.16-cm-outer-diameter swaged sheath of 80%Pt-20%Rh material. The thermocouples extended about 1.7 cm upstream of the water-cooled rake body. These dimensions resulted in a length-diameter ratio that minimized conduction errors. The uncorrected thermocouple readings were taken as true total temperatures. Direct temperature measurements were made to exhaust gas temperatures of 1750 K; at higher conditions the temperature rakes were removed.

Static pressure ports: Two static pressure ports were located in the outer wall of the exhaust annulus or throat, at circumferential positions of 40° and 220° (fig. 12). Two additional static pressure ports were located in the outer wall of the rotating drum. Their circumferential positions referenced to gas sample rake 1 with the center port (port 3) at top dead center are shown in figure 12.

Procedure and Test Conditions

The desired values of combustor inlet pressure, airflow, and fuel-air ratio for any test condition were entered into the facility operation computer file with a given step number. By recalling any step number, the computer controlled the facility valves to give the desired inlet conditions.

The combustion system performance was evaluated at several different modes of operation. A brief description of the operating modes and procedures follows.

Percent of inner-module-row fuel flow to total flow. — Initial tests were conducted to obtain an optimum ratio of inner-module-row fuel flow to total fuel flow (i.e., flow split). The optimum ratio was determined from the span position of the maximum radial temperature and from the pattern factor. Most tests were conducted at an inlet pressure and temperature of 0.79 MPa and 562 K, respectively, and at diffuser inlet Mach numbers of 0.200 and 0.309. The percentage of inner-module-row fuel flow to total flow was varied from 0.40 to 49.9. Selected data points are listed in table II to show examples of the operating conditions and performance results. Exhaust gas thermocouple rakes were used to obtain the performance values, such as exhaust gas temperature, combustion efficiency, and pattern factor.

Ignition operating conditions.—The initial turbine cooling tests in the research facility will be with minimally cooled turbine blades and vanes. Ignition conditions were therefore chosen that would permit repeatable ignition and stable combustion with local gas temperatures less than 1370 K. The nominal conditions, which were not necessarily engine related, were as follows: inlet pressure, 0.79 MPa; inlet air temperature, 560 K; diffuser inlet Mach number, 0.20; and fuel-air ratio, 0.015. Examples of the operating conditions and resultant data are listed in table II.

Idle operating conditions.—Idle is an operating condition that has stable combustion, good efficiency, and enough throughput to operate turbine engines at idle speed. Idle condition can be used before and after test runs to monitor operations and instrumentation. Nominal operating conditions selected for this condition were not intended to be comparable to jet engine nominal inlet conditions and were as follows: inlet pressure, 0.86 MPa; inlet air temperature, 560 K; diffuser inlet Mach number, 0.30; and fuel-air ratio, 0.016. As described under ignition conditions, the maximum local gas temperature was restricted to 1370 K at the turbine inlet. The performance values obtained at idle operating conditions for several data points are given in table II.

Performance determined by exhaust thermocouple data.—Combustor system performance data were obtained from the exhaust gas thermocouple rakes during parametric variation of inlet conditions. Three nominal inlet pressures of 0.65, 0.79, and 0.93 MPa; a nominal inlet air temperature of 560 K; and nominal diffuser inlet Mach numbers of 0.20, 0.25, 0.27, and 0.31 were used for the parametric tests. At any particular inlet air pressure and flow, the fuel-air ratio was varied to about 0.024. For the parametric performance tests of the combustor the maximum local gas temperature was limited to 1750 K in order to minimize any overtemperature damage to the thermocouple rakes during the investigation. Operating conditions and resulting performance values are given in table II.

Performance determined from exhaust gas analysis data.—The exhaust thermocouple rakes were removed in order to determine performance at exhaust gas temperatures greater than 1750 K. These temperatures could damage the thermocouple rakes. During this mode of operation performance was determined from gas analysis data. The inlet air pressures and temperatures were similar to those used during the parametric tests with the exhaust thermocouple rakes installed. However, fuel-air ratios were varied to 0.058. Several test points at parametric operating conditions are listed in table II.

To determine the cooling characteristics of various geometries of turbine blades and vanes, the turbine parametric tests without the exhaust thermocouple rakes were conducted at a constant turbine inlet Mach number of 0.23. For any one turbine inlet pressure, as the fuel-air

ratio was changed, inlet airflow was adjusted to maintain this turbine inlet Mach number. This procedure did not follow actual engine operation. However, the results indicate the effect of individual parameter changes on cooling characteristics of various turbine vane and blade geometries.

Smoke density.—The smoke number of the exhaust gas was determined by diverting part of the gas sample from the gas analysis console to a model 473A engine smoke emission sampler, manufactured by Robert Smith Electric Co., Inc. The stain obtained on number 4 Whatman filter paper was used to determine the smoke number.

Isothermal data.—Inlet air pressure, temperature, and flow were varied (nonburning) to obtain a range of diffuser inlet Mach numbers with which to determine the various combustion system pressure losses as a function of Mach number. The overall system pressure loss included losses from the inlet diffuser, the fuel modules and blocking plate, and the combustor liners. The system pressure loss was determined from inlet and exhaust average total pressures; the fuel-module and blocking-plate loss was calculated from static pressures at the upstream and downstream sides of the blocking plate; the combustor liner loss was computed from annulus average static pressures and exhaust average total pressures.

Results

Effect of Radial Fuel Distribution

The combustion system investigated had two annular rows of fuel modules. To determine an optimum fuel flow ratio between rows, tests were made at various fuel flow splits.

Exhaust radial profile.—The radial profile is the calculated average of all circumferential temperatures at any one radius plotted against the radial position across the annulus (percent span). Figure 13 shows both radial average and maximum individual temperature profile data for fuel flow splits from 0.40 to 49.9 percent of total flow to the inner row of modules. The plotted curves do not represent a required ideal radial profile but only the average test data obtained. The data were obtained at nominal conditions of inlet pressure, 0.79 MPa; inlet air temperature, 562 K; and average exhaust gas temperature, 1172 K. Diffuser inlet Mach numbers of 0.200 and 0.309 were used for most of the tests.

The maximum average radial temperature across the annulus was about 215 kelvins greater than the overall average at the 0.40-percent flow split (fig. 13(a)) and 185 kelvins greater at the 49.9-percent flow split (fig. 13(e)). The peak temperature positions, percent of span from hub to tip, for the two flow splits were about 72 and 32 percent, respectively. The 35.5-percent flow split (fig.

13(d)) gave the least difference, 140 kelvins, between the maximum radial average and the overall average. At this flow split the peak temperature position was about 52 percent. Also shown in figure 13 is the difference between the maximum temperature at any particular radius and the overall average. For the 0.40-percent flow split the difference was 375 kelvins; for the 35.5-percent flow split, 240 kelvins; and for the 49.9-percent flow split, 370 kelvins. Data presented in figure 13 show that increasing the diffuser inlet Mach number from 0.200 to 0.309 had little effect on the exhaust gas profile data, either radial average or radial maximum.

Radial profile data presented in figure 14 were obtained at inlet conditions different from the previously listed conditions: inlet pressure, 0.93 MPa, inlet air temperature, 562 K; and average exhaust gas temperature, 1385 K. The diffuser inlet Mach number was 0.27, and 33.6 percent of the total fuel flow was routed to the inner row of fuel modules. Comparing data shown in figures 14 and 13(d) (similar fuel splits) shows that increasing the overall average exhaust gas temperature resulted in a more center-peaked profile. However, neither the maximum radial average nor the maximum radial temperature increased as much as the overall average. The overall average temperature increased by 213 kelvins; the maximum radial average temperature, by 95 kelvins; and the maximum radial temperature, by 135 kelvins.

The combustion efficiency for the data presented in figure 14 was about 98 percent as compared with 94 percent for the data shown in figure 13(d). The change in profile and combustion efficiency was mostly due to the increase in the differential temperature rather than the small increase in the combustion pressure.

Combustion efficiency and pattern factor.—Combustion efficiency is a calculated thermal efficiency either obtained from exhaust gas thermocouples or calculated from analysis of exhaust gas products when the thermocouple rakes were removed. Pattern factor is defined as follows:

$$\frac{T_{ex,max} - T_{ex,av}}{T_{ex,av} - T_{in,av}}$$

where $T_{ex,av}$ is maximum local exhaust temperature, $T_{ex,av}$ is average exhaust temperature, and $T_{in,av}$ is average inlet air temperature.

Combustion efficiency and pattern factor data obtained while testing variation in fuel flow splits to the fuel modules are shown in figure 15. Variation in combustion efficiency was minimal for the full range of fuel flow splits (fig. 15(a)), varying from 96.5 percent for the 0.40-percent split to 93.5 percent for the 49.9-percent split. However, pattern factor changed considerably over

the fuel split range, varying from about 0.35 to 0.66 (fig. 15(b)). A pattern factor of 0.66 was obtained at the 0.40-percent split; 0.35, at the 34.8-percent split; and 0.60, at the 49.9-percent split. The minimum pattern factor, which is desirable from a turbine durability standpoint, was about 0.35 and occurred at a flow split of 34.8 percent.

At any one operating condition, changes in the fuel flow split between the inner and outer rows of fuel modules did affect the radial profile and the radial profile peak position. Changing the fuel flow split can be used to adjust the exhaust gas radial profile for optimum turbine operation. Varying the flow split from 0.40 percent to 49.9 percent had only minimal effect on combustion efficiency but a considerable effect on pattern factor. Since the minimum pattern factor, obtained from tests at various fuel flow splits, was about 0.35 at 34.8-percent split, most combustion tests were conducted at flow splits of 30 to 40 percent.

Ignition Operating Conditions

The low fuel flows during ignition resulted in very low fuel-module injection pressure. To determine if this low injection pressure would degrade combustor ignition, several tests were conducted. Criteria were repeatable ignition, stable combustion, and a maximum local gas temperature less than 1370 K. As described in the section Procedure and Test Conditions, 1370 K was a maximum local turbine inlet temperature limit imposed because minimally cooled turbine blades and vanes will be used for initial turbine cooling investigations.

Results of tests at 0.79-MPa inlet pressure and 556 K inlet air temperature are given in figure 16. Average exhaust gas temperatures and maximum local exhaust gas temperatures are shown in figure 16(a) as functions of fuel-air ratio. With an average exhaust gas temperature of about 1080 to 1100 K the maximum local temperature was 1290 to 1330 K. The combustion efficiency range at these conditions was 91 to 93 percent (fig. 16(b)). At these operating conditions ignition was repeatable, combustion was stable, and the maximum local exhaust gas temperature was less than 1370 K.

Idle Operating Conditions

The turbine must be accelerated to an idle condition after a successful ignition. At this operating condition combustion should be stable, performance repeatable, and the combustor output (mass flow and temperature) sufficient to drive the turbine to its idle speed. Data taken at the idle condition before and after test runs permitted a check of instrumentation and research equipment performance. Idle conditions chosen were 0.86-MPa inlet pressure and 560 K inlet air temperature. Results showing repeatability, average and maximum local gas

temperatures, and combustion efficiency are given in figure 17. Operation over fuel-air ratios of 0.0158 to 0.0168 gave stable combustion and repeatable maximum local gas temperatures less than 1370 K.

Combustor Performance Obtained from Both Thermocouple and Gas Analysis Data

Operation of the combustor to exhaust gas temperatures in excess of 1750 K required that the exhaust gas thermocouple rakes be removed. To compare performance data from gas analysis with those obtained by thermocouple rakes, tests were conducted at conditions where both types of instrumentation were used.

Conditions chosen were inlet pressures of 0.65, 0.79, and 0.93 MPa, an inlet air temperature of 562 K, and diffuser inlet Mach numbers of 0.200, 0.247, 0.274, and 0.309. Fuel-air ratios varied from 0.014 to 0.0238. The highest fuel-air ratio was limited by the imposed maximum local exhaust gas temperature of 1750 K.

Data obtained at the 0.65-MPa inlet pressure are given in figure 18. As described in the section Research Instrumentation, the five sample ports of a gas sample rake were ducted to a common plenum inside the rake. Therefore the maximum local exhaust gas temperature could not be calculated.

Comparing figures 18(a-1) and (b-1) shows good agreement of the average exhaust gas temperatures obtained by the two types of instrumentation. Combustion efficiencies (figs. 18(a-2) and (b-2)) were slightly higher by gas analysis calculations (96.5 to 95.0 at 0.02 fuel-air ratio). At the lower fuel-air ratios an increase in diffuser inlet Mach number caused a slight decrease in combustion efficiency.

Figures 19 and 20 present data at inlet pressures of 0.79 and 0.93 MPa, respectively. Again, there was good agreement of the exhaust gas average temperature measurements by the two types of instrumentation. At the 0.79-MPa inlet pressure the combustion efficiency values obtained by gas analysis were slightly higher than those obtained by thermocouple measurements (figs. 19(a-2) and (b-2)). However, at the 0.93-MPa inlet pressure the agreement was good (figs. 20(a-2) and (b-2)). At the higher inlet pressure, combustion efficiency approached 98.5 percent at the high fuel-air ratios. The combustion efficiency values calculated from gas analysis showed a somewhat wider spread for variation in diffuser inlet Mach number than those calculated from thermocouple data. The data obtained at the three inlet pressures indicate that the combustor performance calculated from gas analysis data is comparable to that calculated from thermocouple data. This close agreement for the same operating condition indicates that the particular type of gas sample rakes used collected a representative gas sample and that the sample

line properly conditioned the sample for the analysis consoles.

Combustor Performance Obtained From Gas Analysis Data

Removal of the exhaust thermocouple rakes made it possible to operate to a much higher exhaust gas temperature. Data were obtained at fuel-air ratios to 0.058 as compared with the maximum fuel-air ratio of 0.0238 tested with exhaust thermocouple rakes installed. Inlet pressures were 0.65, 0.80, and 0.94 MPa and the inlet air temperature was about 560 K. As described in the section Procedure and Test Conditions the parametric test procedure used at the higher fuel-air ratio was different from that used when the exhaust gas temperature was limited because of installed thermocouple rakes. For these tests the turbine inlet Mach number was held constant at 0.23 by adjusting inlet airflow, and this resulted in varying values of diffuser inlet Mach number. This procedure is different from actual engine operation, but it does permit better evaluation of individual parameter effects on the cooling characteristics of various geometries of turbine blades and vanes. However, it can cause combustor liner cooling problems. The decrease in airflow required to maintain the constant turbine inlet Mach number, because of increasing turbine inlet gas temperatures, caused a decrease in combustor differential pressure and therefore a decrease in the liner film cooling air. This type of operation resulted in the lowest film-cooling-air flow rates at the highest exhaust gas temperature. For the particular test conditions the variation in inlet airflow rate gave diffuser inlet Mach numbers that ranged from 0.22 to 0.36.

Data at 0.65-MPa pressure are presented in figure 21. Average exhaust gas temperature and combustion efficiency are plotted as a function of fuel-air ratio in figures 21(a) and (b). An average exhaust gas temperature of 2135 K was obtained at a fuel-air ratio of 0.058. The data show the variation in the diffuser inlet Mach number, the lowest value being at the highest gas temperature. The maximum combustion efficiency of 98.5 percent occurred at a fuel-air ratio of about 0.033 and decreased to 91 percent at a fuel-air ratio of 0.058. The decrease in combustion efficiency with increasing fuel-air ratio was mainly due to the increasingly high fuel-air ratios in the wake of the fuel modules. For example, at an overall fuel-air ratio of 0.058 the ratio just downstream of the fuel modules was above stoichiometric. Because the combustor liner does not have dilution air penetration holes, only film cooling air enters the combustion zone downstream of the fuel modules. Therefore mixing for combustion purposes is dominated by airflow entering through the module

swirlers. At high fuel-air ratios combustion efficiency begins to fall because there is insufficient mixing and not enough time available to complete the combustion process. Gas analysis data showed that the levels of incomplete combustion products, mostly carbon monoxide, increased rapidly at fuel-air ratios above about 0.04. Levels of unburned hydrocarbons also increased but did not contribute significantly to combustion inefficiency.

Figure 21(c) presents maximum liner metal average differential temperature (32 thermocouples, 16 on each liner) and the maximum (of the 32 thermocouples) local liner temperature minus the inlet air temperature. The maximum liner metal average differential temperature was about 375 kelvins above the inlet air temperature; the maximum local liner temperature was 575 kelvins above the inlet air temperature. This corresponds to a metal temperature of 1137 K, which is less than the 1260 K design limiting temperature. Generally, the maximum local liner differential temperatures were indicated by thermocouples on the seventh or last liner panel, and the maximum temperatures were not always at the same locations.

The narrow range of inlet pressures had little effect on average exhaust gas temperature or combustion efficiency. The maximum gas temperature was 2100 to 2135 K at fuel-air ratios of 0.056 to 0.058, and the maximum combustion efficiency was 98.5 percent at a fuel-air ratio of 0.033 (figs. 21 to 23). Average liner metal differential temperatures were quite similar for any specified fuel-air ratio at the various inlet pressures (figs. 21(c), 22(c), and 23(c)). However, the maximum liner metal differential temperature was much higher at the lowest inlet pressure especially for fuel-air ratios from 0.02 to 0.055. For example, at a fuel-air ratio of 0.033 the maximum differential temperature was about 460 kelvins above the inlet air temperature at 0.65 MPa and 345 kelvins above at 0.94 MPa. Thus for comparable diffuser inlet Mach numbers and similar fuel-air ratios, peak liner metal temperatures decreased at the highest inlet pressures. This decrease indicates that the maximum local liner temperatures may not become excessive as the combustion pressure is increased to much higher values.

Smoke Density

Smoke density data are given in figure 24 as a function of fuel-air ratio. The figure includes test results at three inlet pressures and four diffuser inlet Mach numbers. Inlet air temperature and turbine inlet Mach number were constant for the tests.

The density numbers were below 10 to a fuel-air ratio of 0.030 and then increased to about 20 as the fuel-air ratio was increased to 0.036. The smoke numbers increased much more rapidly after a fuel-air ratio of

0.034, attaining a value of 95 at a fuel-air ratio of 0.057. This abrupt change in slope can occur when the mixture ratio exiting from the fuel modules becomes greater than stoichiometric.

Smoke number values reported in reference 7, obtained from a combustion system similar to the one used for the tests reported herein, did not show the abrupt increase until an overall fuel-air ratio of about 0.052. Two differences, one in hardware and one in test procedure, probably account for the much lower smoke numbers reported in reference 7. First, the fuel-module blocking plate (ref. 7) had a much smaller flowthrough area, which would cause more air to flow through the fuel modules at similar conditions. Consequently, the fuel-module mixture ratio would not be in excess of the stoichiometric value until a larger overall fuel-air ratio was reached. Second, the test procedure used in reference 7 maintained a constant diffuser inlet Mach number or reference velocity with increasing fuel-air ratio. However, the test procedure for the data reported herein maintained a constant turbine inlet Mach number with increasing fuel-air ratio. At constant pressure this resulted in a decrease in reference velocity. Data presented in reference 8 show substantial increases in smoke number as reference velocity is decreased.

The test data reported herein indicate that the narrow range of combustion pressures tested had only a small effect on smoke number. However, large increases in combustion pressure can cause a considerable increase in smoke number. As shown in references 8 and 9 the magnitude of increased smoke number with an increase in combustion pressure depends on fuel injector type and combustor geometry.

Isothermal Tests

Inlet air temperature, pressure, and flow were varied without combustion to obtain the isothermal pressure loss of the combustor, which includes the inlet diffuser, the fuel modules and blocking plate, and the combustor liners. Examples of the various inlet conditions and results are given in table III. The combustion system overall pressure loss is defined as

$$\frac{P_{in,av} - P_{ex,av}}{P_{in,av}}$$

where $P_{in,av}$ is average inlet total pressure and $P_{ex,av}$ is average exhaust total pressure. The pressure loss across the fuel modules and blocking plate is defined as

$$\frac{P_{bu,av} - P_{bd,av}}{P_{in,av}}$$

where $p_{bu,av}$ is blocking-plate upstream average static pressure and $p_{bd,av}$ is blocking-plate downstream average static pressure. Since the inlet air Mach numbers at the upstream and downstream sides of the fuel-module blocking plate are low, static pressures can be used as total pressures. The liner pressure loss is defined as

$$\frac{p_{l,av} - P_{ex,av}}{P_{in,av}}$$

where $p_{l,av}$ is liner annulus average static pressure. The combustor liners do not have air dilution holes; so only film cooling air flows in the combustor annuli. Since the Mach number is low, annulus static pressures can be used as total pressures.

Data in figure 25 show pressure loss as a function of diffuser inlet Mach number. The diffuser inlet Mach numbers range from 0.190 to 0.370; this results in system losses from 1.75 to 7.3 percent, fuel-module and blocking-plate losses from 0.80 to 3.5 percent, and combustor liner losses from 0.25 to 1.5 percent. The combustor liner pressure losses at the low Mach numbers indicate low liner cooling airflows. At a diffuser inlet Mach number of 0.35, system loss is 6.5 percent, fuel-module and blocking-plate loss is 2.6 percent, and combustor liner loss is 1.1 percent. This indicates that there is about a 2.8-percent loss in the diffuser and in the turning of the flow path leading to the liner annuli.

Discussion

The combustion system used to obtain the reported data was designed to furnish inlet pressures to 4.05 MPa, inlet air temperatures to 900 K, and exhaust gas temperatures to 2480 K to turbine cooling research equipment. The design emphasized aircraft engine hardware geometries rather than boilerplate ground slave equipment. The first series of tests were conducted at inlet pressures to 0.94 MPa, a nominal inlet air temperature of 560 K, and exhaust gas temperatures to 2135 K (0.058 fuel-air ratio).

Combustion efficiency varied considerably over the fuel-air ratio range investigated—increasing from 91 percent at 0.016 to 98.5 percent at 0.033 and then decreasing to about 90 percent as the fuel-air ratio was increased to 0.058. The fuel injection modules were designed to give good atomization at high fuel flow and airflow rates and with high inlet air temperatures, as is typical of engine operation at pressures of 2.03 to 4.05 MPa. Therefore operation at low fuel-air ratios and with low inlet air temperature (equal or less than the fuel ASTM final boiling point) results in a fuel vaporization rate that is low enough to be the major time step in the

combustion process. As a result some of the fuel does not have time for complete combustion. As described in the Procedure and Test Conditions section the turbine cooling parametric tests will use a constant turbine inlet Mach number for all conditions. So, for a constant inlet pressure, as the exhaust gas temperature was increased, the airflow rate was decreased. This decrease in airflow resulted in lower pressure loss and hence less mixing of the air within the combustor. Continued decrease in airflow rates with increasing fuel-air ratio will result in the mixing time (length) becoming a major time step in the combustion process, finally resulting in decreased combustion efficiencies.

The test mode of operation will decrease cooling airflow to the liner during the time when the heat load rate to the liner is increasing. For the tests reported, the maximum local liner metal temperature, at the highest exhaust gas temperature, was 125 kelvins below the liner design maximum temperature. A small decrease in the maximum local liner metal temperature was obtained with an increase in combustion pressure over the narrow pressure range investigated. Therefore maximum liner temperatures may not become excessive as the combustion pressure is increased.

The combustion system used for these investigations had two annular rows of fuel modules mounted in the same plane. Since the combustor liner did not have any air dilution holes, the fuel flow split between the inner and outer rows of fuel modules was varied in order to achieve the desired radial profile. The position of the peak profile average temperature across the annulus could be varied from about 32 to 72 percent of span, hub to tip, by changes in the flow split. Although these changes had only a minor effect on combustion efficiency, there was a large change in pattern factor. The performance tests were conducted with a flow split that produced the lowest range of pattern factors.

A combustor designed to furnish a high-temperature, high-pressure environment to turbine cooling research equipment must also be capable of repeatable ignition and stable combustion and able to accelerate the turbine to a required idle rpm condition. Since the preliminary turbine cooling investigations will be with minimally cooled turbine blades and vanes, the combustion system must supply an environment with limited maximum local temperatures at ignition and furnish enough throughput to accelerate the turbine to idle rpm, again with the limited maximum local exhaust gas temperature. The operating conditions selected for ignition and idle operation are not intended to be comparable to jet engine nominal inlet conditions. The combustion system, operating at the selected ignition conditions, did show repeatable ignition, stable combustion, and maximum local exhaust gas temperatures less than the limiting value imposed for protection of the minimally cooled turbine blades and vanes. The combustion system, when

operated at the conditions selected for idle rpm operation, showed stable combustion, good efficiency, and sufficient throughput to accelerate the turbine to idle rpm. Also maximum local temperatures were less than the imposed limiting value.

Operation of the combustion system at high fuel-air ratios resulted in high smoke density numbers, much higher than data reported in reference 7 for a similar combustion system. A hardware difference that could have caused some increase in smoke density is the more open flow area in the fuel-module blocking plate of the present combustor. Since fuel atomization in the fuel modules depends greatly on airflow through the module, the more open flow area of the blocking plate will result in decreased airflow through the fuel modules and therefore less fuel atomization (larger droplets). Larger fuel droplets are conducive to greater smoke density. Another possible reason for the increased smoke density is the constant exit Mach number mode of combustor operation. Operation at constant exit Mach number necessitates reducing the reference velocity. As reported in reference 8, reductions in reference velocity cause an increase in smoke density. The mode of operation for tests reported in reference 7 kept a constant diffuser inlet Mach number and airflow rate through the fuel modules with an increase in exhaust gas temperature and hence lower smoke numbers at high fuel-air ratios.

Summary of Results

A combustion system was designed to operate at an inlet pressure of 4.05 MPa (40 atm), an inlet air temperature of 900 K, and an exhaust gas temperature of 2480 K. A preliminary investigation of this combustor was conducted at nominal conditions of pressure, to 0.94 MPa (9 atm), inlet air temperature, 560 K; and exhaust gas temperature, to 2135 K. The following results were obtained:

1. Combustion efficiency varied from about 90 percent at a fuel-air ratio of 0.016 (1100 K exhaust gas temperature) to 98.5 percent at a fuel-air ratio of 0.033 (1650 K) and then decreased to 91 percent as fuel-air ratio was further increased to about 0.055 (2090 K).

2. At an exhaust gas temperature of 2090 K (0.055 fuel-air ratio), the average of all combustor liner thermocouples indicated an average metal temperature 365 kelvins greater than the inlet air temperature. The maximum metal temperature differential at the 0.055 fuel-air ratio was 565 kelvins at 0.65-MPa inlet pressure, 535 kelvins at 0.80 MPa, and 505 kelvins at 0.94 MPa. Thus maximum liner temperatures may not become excessive as the combustor pressure is increased.

3. A fuel flow rate to the inner row of fuel modules of about 35 percent of total fuel flow gave a minimum

pattern factor of about 0.35 and a maximum average radial temperature position at 52 percent of span, hub to tip.

4. Combustor ignition was reproducible and stable with the maximum local exhaust gas temperature less than 1340 K at the following nominal ignition conditions: 0.79-MPa inlet pressure, 556 K inlet air temperature, 11.2-kg/sec airflow, and 0.0153 fuel-air ratio.

5. An idle operating condition of 0.86-MPa air pressure, 560 K air temperature, 17.98-kg/sec airflow, 0.0157 fuel-air ratio gave stable combustion, an average exhaust gas temperature of 1050 K, and maximum local temperature less than 1290 K. Combustion efficiency was about 93 percent.

6. Combustion efficiency and average exhaust gas temperature calculated from both thermocouple and gas analysis data showed good agreement at fuel-air ratios from 0.014 to 0.023. The effect on combustion efficiency, with variation of diffuser inlet Mach number, was somewhat greater for the gas analysis data than for the thermocouple data. The thermocouples were of a nonaspirating type, and the actual readings were taken as true total temperatures.

7. Smoke numbers of 4, 20, 45, and 95 were obtained at fuel-air ratios of 0.016, 0.036, 0.045, and 0.057, respectively. Smoke numbers greater than a value of 25 are considered to be in the visible region.

8. The combustion system isothermal pressure loss was about 6.5 percent of the inlet total pressure at a diffuser inlet Mach number of 0.35; the combustor liner loss was about 1.1 percent. The low combustor liner loss compared with the overall system loss indicates a considerable loss in the diffuser and in the turning of the flow path leading to the liner annuli.

National Aeronautics and Space Administration
Lewis Research Center
Cleveland, Ohio, April 18, 1983

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TABLE I. - FILM-COOLING-AIR HOLES OF COMBUSTOR LINERS

Panel	Number of film holes per panel	Diameter of film holes, cm	Total film-hole area per panel, A, cm ²	Calibration value of Cd ^a	Calibration value of ACd ^b
Inner liner					
1 (up-stream)	142	0.208	4.838	-----	-----
2	132	.237	5.847	-----	-----
3	132	.264	7.234	-----	-----
4	132	.264	7.234	-----	-----
5	142	.278	8.612	-----	-----
6	162	.237	7.176	-----	-----
7 (down-stream)	178	.218	6.671	-----	-----
All panel holes unblocked			47.613 cm ² /liner	0.00344	0.1638
Outer liner					
1 (up-stream)	238	0.208	8.109	-----	-----
2	252	.208	8.586	-----	-----
3	258	.249	12.555	-----	-----
4	228	.249	11.095	-----	-----
5	220	.258	11.484	-----	-----
6	208	.226	8.348	-----	-----
7 (down-stream)	196	.244	9.165	-----	-----
All panel holes unblocked			69.342 cm ² /liner	0.00368	0.2552
All panel holes unblocked, both liners			116.955 cm ² /both liners	0.003625	0.4240

^aValue of flow coefficient Cd reported in ref. 4 for a combustor liner of similar geometry, but different film-hole areas, with all panel holes unblocked.

^b $W = ACd \sqrt{\rho \Delta P}$, where W is film-cooling-air flow in kilograms per second; A is area of unblocked film-cooling-air holes in square centimeters; Cd is flow coefficient; ρ is density of film cooling air entering liner in kilograms per cubic meter; and ΔP is pressure differential across liner in kilopascals.

TABLE II. - TEST CONDITIONS AND EXPERIMENTAL DATA

Combustor inlet conditions							Combustor operation results											
Total pressure, MPa	Total temperature, K	Air flow, kg/sec	Mach number		Fuel-air ratio	Fuel module pressure differential, kPa		Exhaust temperature, mass average, K	Combustion efficiency, mass average, percent	Exhaust temperature, gas analysis, K	Combustion efficiency, gas analysis, percent	Exhaust temperature, maximum local, K	Mach number, turbine inlet	System total pressure loss, percent	Smoke number	Fuel flow, inner row to total, percent	Temperature, liner average, K	Temperature, liner maximum local, K
			Diffuser inlet	Reference		Inner row	Outer row											
Variation in ratio of fuel, inner to outer row of fuel modules																		
0.792	555	11.23	0.201	0.0195	0.0152	14.7	160	1099	96.6	----	----	1486	0.129	2.2	----	0.40	676	874
.791	563	16.66	.309	.0292	.0176	51.6	258	1165	93.8	----	----	1442	.208	5.1	----	25.8	676	884
.790	561	16.65	.309	.0292	.0175	55.4	230	1168	94.7	1171	95.1	1417	.209	5.2	----	30.1	674	853
.791	563	11.13	.200	.0195	.0176	33.9	92.7	1171	94.8	----	----	1405	.133	2.3	----	35.4	688	865
.792	564	16.66	.309	.0292	.0176	74.6	200	1168	94.2	----	----	1401	.208	5.2	----	35.5	676	862
.791	561	16.66	.309	.0292	.0175	127	130	1161	93.6	1167	94.4	1507	.208	5.1	----	49.8	668	815
.934	561	17.65	.274	.0262	.0237	119	400	1379	97.4	1386	98.3	1747	.203	4.2	----	33.6	708	932
Ignition operating conditions																		
0.794	563	11.21	0.201	0.0196	0.0150	26.7	74.0	1077	92.1	----	----	1298	0.127	2.1	----	34.7	671	798
.792	554	11.21	.200	.0195	.0153	28.1	69.2	1077	92.0	----	----	1310	.128	2.1	----	35.8	662	783
.792	561	11.13	.200	.0195	.0157	33.4	74.6	1105	93.7	1122	96.7	1323	.129	2.1	----	37.3	672	801
.792	552	11.22	.200	.0195	.0153	33.2	63.2	1076	92.4	----	----	1297	.128	2.1	----	41.2	660	800
.792	558	11.22	.201	.0196	.0151	15.7	84.0	1071	91.6	----	----	1335	.128	2.3	----	25.9	663	813
.792	559	11.23	.201	.0196	.0150	26.7	74.4	1082	93.6	----	----	1291	.127	2.3	----	34.2	667	824
Idle operating conditions																		
0.864	559	17.96	0.303	0.0287	0.0167	81.5	198	1136	93.8	1151	96.2	1365	0.202	5.0	----	38.4	668	818
.863	559	17.97	.304	.0288	.0157	83.1	162	1097	92.4	1101	93.2	1327	.198	5.0	----	40.9	663	814
.863	560	17.98	.304	.0288	.0157	54.3	185	1099	93.0	1103	93.7	1311	.198	5.1	----	34.1	666	816
.862	562	17.96	.305	.0289	.0157	54.1	185	1101	93.0	1104	93.5	1311	.198	4.9	----	34.5	666	816
.863	561	17.98	.305	.0289	.0157	67.1	193	1101	93.5	1101	93.4	1308	.199	5.0	----	33.6	664	813
Performance determined by exhaust thermocouple data																		
0.649	561	11.14	0.247	0.0237	0.0211	56.0	126	1281	95.1	----	----	1555	0.175	3.5	----	36.7	694	879
.648	562	11.13	.247	.0238	.0176	45.8	82.3	1167	94.1	1176	95.6	1460	.165	3.3	----	40.5	678	851
.648	562	11.14	.247	.0238	.0142	29.0	53.7	1036	89.3	1040	90.1	1275	.154	3.6	----	41.1	655	780
.648	562	11.14	.247	.0238	.0157	41.6	70.2	1100	92.4	1108	93.9	1343	.160	3.4	----	39.4	666	818
.648	561	12.25	.274	.0261	.0157	44.9	91.6	1096	92.1	----	----	1333	.177	4.2	----	37.5	664	806
.647	560	12.26	.274	.0262	.0175	53.8	97.9	1160	93.4	1174	95.5	1435	.183	4.1	----	40.4	673	841
.648	562	12.27	.274	.0262	.0211	74.7	140	1281	94.9	1296	96.8	1586	.195	4.2	----	39.1	693	910
.648	562	12.25	.274	.0262	.0142	43.2	71.1	1029	88.0	1044	90.8	1259	.171	4.1	----	40.1	653	777
.647	561	13.60	.308	.0291	.0142	38.3	99.9	1024	83.7	1033	89.0	1245	.193	5.2	----	34.9	652	780
.646	562	13.60	.309	.0291	.0157	39.2	115	1088	90.4	1100	92.6	1314	.200	5.1	----	35.0	664	821
.647	562	13.60	.308	.0291	.0175	47.2	140	1158	93.0	1170	95.0	1403	.207	5.2	----	34.5	673	839
.647	562	13.60	.308	.0291	.0210	67.7	198	1279	95.1	1296	97.4	1561	.220	5.2	----	34.4	690	908
.793	566	11.14	.201	.0195	.0175	43.1	104	1177	95.5	1190	97.5	1389	.133	2.0	----	34.5	693	857
.794	566	11.14	.201	.0195	.0150	34.9	80.9	1083	92.9	1096	95.3	1289	.127	2.0	----	34.6	674	811
.794	566	11.13	.200	.0195	.0211	37.0	127	1291	95.7	1313	98.6	1565	.141	2.2	----	33.9	716	909
.791	564	11.13	.200	.0195	.0197	39.1	117	1255	96.9	1265	98.2	1530	.139	2.2	----	34.6	704	876
.792	563	11.15	.200	.0195	.0238	53.2	163	1380	97.0	1395	98.8	1730	.147	2.2	----	34.7	731	935
.791	562	14.94	.274	.0262	.0142	47.8	109	1041	90.8	1063	95.0	1255	.172	4.0	----	36.4	657	780
.793	566	14.94	.274	.0262	.0176	59.6	170	1175	94.8	1186	96.6	1404	.184	4.1	----	34.3	683	846
.792	563	14.96	.274	.0262	.0197	71.1	207	1252	96.6	----	----	1521	.192	4.1	----	34.0	687	868
.792	563	14.94	.274	.0262	.0237	93.1	285	1388	98.2	1389	98.3	1742	.204	4.2	----	34.2	707	916
.792	562	16.67	.308	.0291	.0141	49.4	133	1035	89.7	1036	89.9	1242	.194	5.1	----	36.7	656	778
.793	567	16.66	.309	.0292	.0210	95.1	288	1293	96.3	1305	97.9	1589	.221	5.2	----	34.0	697	896
.792	562	16.67	.309	.0292	.0174	73.8	191	1165	94.4	1171	95.4	1390	.208	5.0	----	36.1	674	827
.792	564	16.67	.309	.0292	.0236	116	353	1382	97.6	1385	98.0	1734	.231	5.2	----	34.4	703	910
.934	564	16.10	.249	.0240	.0141	39.8	116	1056	93.5	1056	93.5	1238	.156	3.4	----	35.7	665	795
.933	564	16.07	.248	.0239	.0175	58.6	172	1180	96.4	1184	97.0	1402	.167	3.4	----	35.6	684	845
.933	562	16.11	.249	.0240	.0209	84.8	242	1296	97.9	1298	98.1	1603	.177	3.4	----	36.6	700	901
.933	563	16.09	.248	.0239	.0236	108	305	1383	98.2	1386	98.6	1738	.184	3.4	----	36.4	713	930
.933	562	17.65	.274	.0262	.0175	77.6	211	1176	96.0	1178	96.4	1417	.185	4.1	----	35.3	678	843
.934	564	17.67	.275	.0262	.0196	90.9	274	1256	97.5	1256	97.5	1538	.192	4.1	----	34.4	691	865
.934	564	17.63	.274	.0262	.0142	55.9	153	1055	92.9	1053	92.6	1260	.173	4.0	----	35.0	661	788
.933	563	17.68	.275	.0263	.0235	124	379	1385	98.6	1383	98.3	1697	.204	4.2	----	35.4	710	935
.933	562	19.68	.309	.0292	.0197	114	324	1249	96.7	1250	96.8	1528	.217	5.2	----	35.6	684	870
.934	564	19.68	.310	.0293	.0141	64.8	178	1043	91.0	1040	90.4	1219	.195	5.1	----	35.5	659	785
.933	564	19.72	.311	.0294	.0235	150	467	1384	98.4	1381	98.0	1748	.232	5.2	----	35.1	705	921
.932	563	19.66	.309	.0293	.0209	132	362	1303	98.7	1291	97.2	1600	.223	5.1	----	36.6	691	892
Performance determined by gas analysis data																		
0.649	563	10.19	0.225	0.0218	0.0500	201	546	----	----	2006	92.5	----	0.218	3.2	----	37.1	896	1068
.650	563	10.17	.224	.0217	.0578	260	717	----	----	2137	91.1	----	.232	3.9	----	36.7	937	1137
.649	562	11.11	.246	.0237	.0435	178	483	----	----	1878	94.0	----	.227	3.4	----	36.8	823	984
.649	563	11.02	.244	.0235	.0437	176	499	----	----	1888	94.3	----	.224	2.7	71	36.5	821	952
.649	561	12.84	.288	.0274	.0344	137	426	----	----	1676	98.6	----	.246	4.8	10.8	35.0	769	1029
.650	564	12.20	.272	.0260	.0341	131	367	----	----	1687	98.0	----	.229	3.5	6.9	35.8	744	846
.651	561	13.62	.306	.0289	.0247	90.3	255	----	----	1415	98.0	----	.234	5.0	----	35.2	700	811
.649	561	13.60	.307	.0290	.0247	86.8	264	----	----	1411	98.0	----	.234	5.0	----	35.0	699	813
.651	561	15.50	.354	.0329	.0162	63.2	153	----	----	1104	91.1	----	.239	6.6	----			

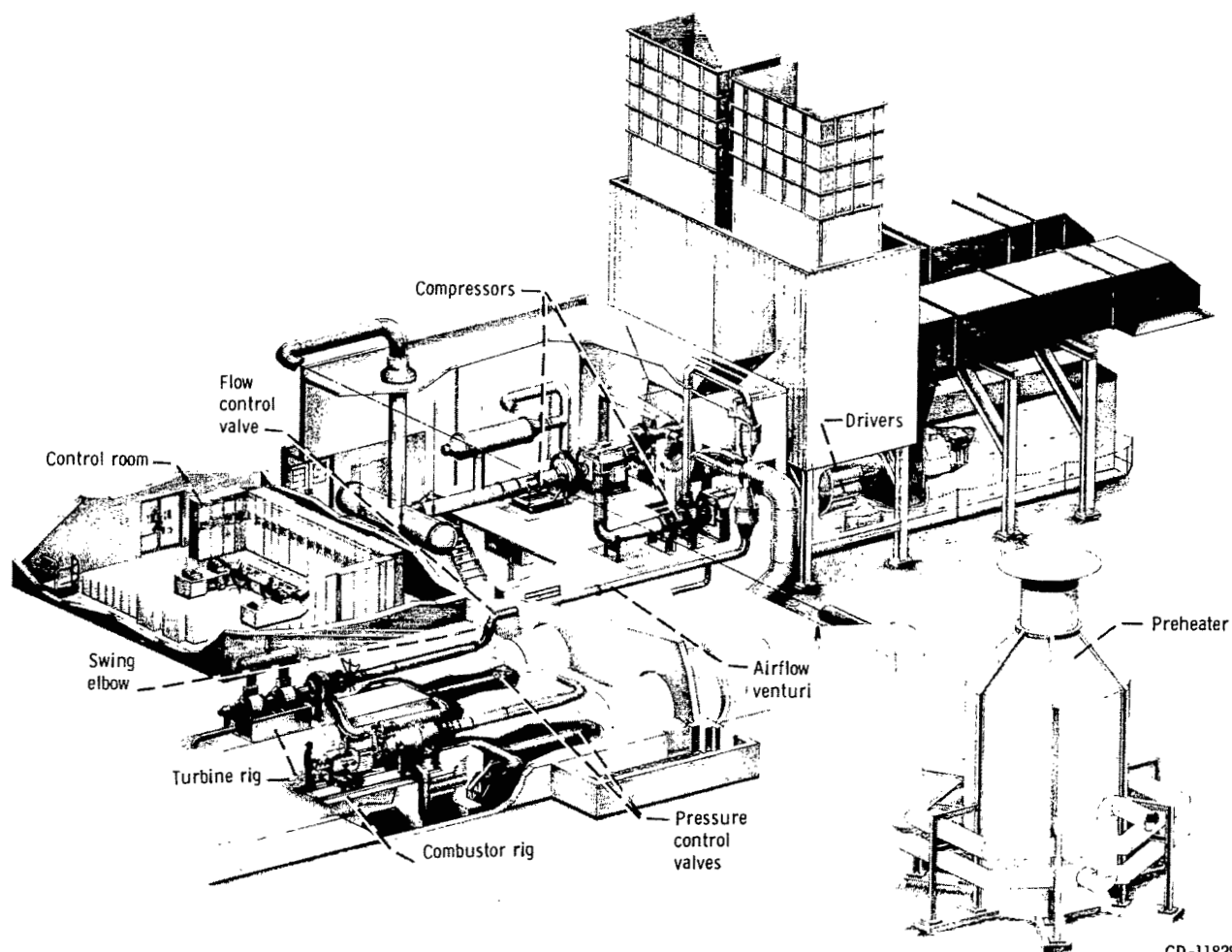
TABLE II. - Concluded.

Combustor inlet conditions								Combustor operation results										
Total pressure, MPa	Total temperature, K	Air flow, kg/sec	Mach number		Fuel-air ratio	Fuel module pressure differential, kPa		Exhaust temperature, mass average, K	Combustion efficiency, mass average, percent	Exhaust temperature, gas analysis, K	Combustion efficiency, gas analysis, percent	Exhaust temperature, maximum local, K	Mach number, turbine inlet	System total pressure loss, percent	Smoke number	Fuel flow, inner row to total, percent	Temperature, liner average, K	Temperature, liner maximum local, K
			Diffuser inlet	Reference		Inner row	Outer row											
Exhaust smoke density data																		
0.795	563	18.99	0.357	0.0331	0.0160	77.2	210	----	----	1123	94.8	----	0.239	6.3	2.4	36.0	660	743
.798	563	16.65	.306	.0289	.0246	126	367	----	----	1418	99.6	----	.233	5.2	3.2	35.5	702	784
.797	564	14.93	.272	.0260	.0335	179	536	----	----	1670	98.0	----	.229	4.3	12.6	36.1	750	849
.798	564	13.59	.245	.0236	.0429	252	721	----	----	1876	94.9	----	.225	3.7	68	36.7	829	946
Isothermal data (nonburning)																		
0.729	436	17.94	0.318	0.0298	-----	-----	-----	-----	-----	-----	-----	-----	0.140	5.3	-----	-----	438	440
.727	549	17.97	.366	.0338	-----	-----	-----	-----	-----	-----	-----	-----	.161	6.4	-----	-----	547	563
.796	552	11.26	.199	.0194	-----	-----	-----	-----	-----	-----	-----	-----	.087	1.8	-----	-----	550	562
.725	545	17.96	.365	.0337	-----	-----	-----	-----	-----	-----	-----	-----	.161	6.4	-----	-----	540	543
.729	509	17.97	.349	.0324	-----	-----	-----	-----	-----	-----	-----	-----	.155	6.4	-----	-----	503	507
.643	282	15.52	.243	.0233	-----	-----	-----	-----	-----	-----	-----	-----	.106	3.0	-----	-----	282	283
.730	412	17.91	.307	.0289	-----	-----	-----	-----	-----	-----	-----	-----	.135	5.0	-----	-----	404	410
.732	462	17.87	.326	.0305	-----	-----	-----	-----	-----	-----	-----	-----	.143	5.2	-----	-----	453	458

TABLE III. - ISOTHERMAL TEST CONDITIONS

Combustor inlet			Diffuser inlet Mach number	Average pressure differential across liners, kPa	Flow rate, ^a kg/sec			Flow, percent of combustor total	
Total pressure, MPa	Airflow rate, kg/sec	Total temperature, K			Inner liner	Outer liner	Both liners	Inner liner	Outer liner
0.729	17.94	436	0.318	6.343	0.974	1.517	2.520	5.4	8.5
.796	11.26	552	.199	2.151	.534	.832	1.383	4.7	7.4
.643	15.53	282	.243	2.958	.784	1.222	2.630	5.1	7.9
.732	17.87	462	.326	6.736	.977	1.523	2.530	5.5	8.5

^aCalculated from film-cooling-air flow equations previously noted in table I, and isothermal ΔP values from fig. 25.



CD-11828-11

Figure 1. - High-pressure, high-temperature facility.

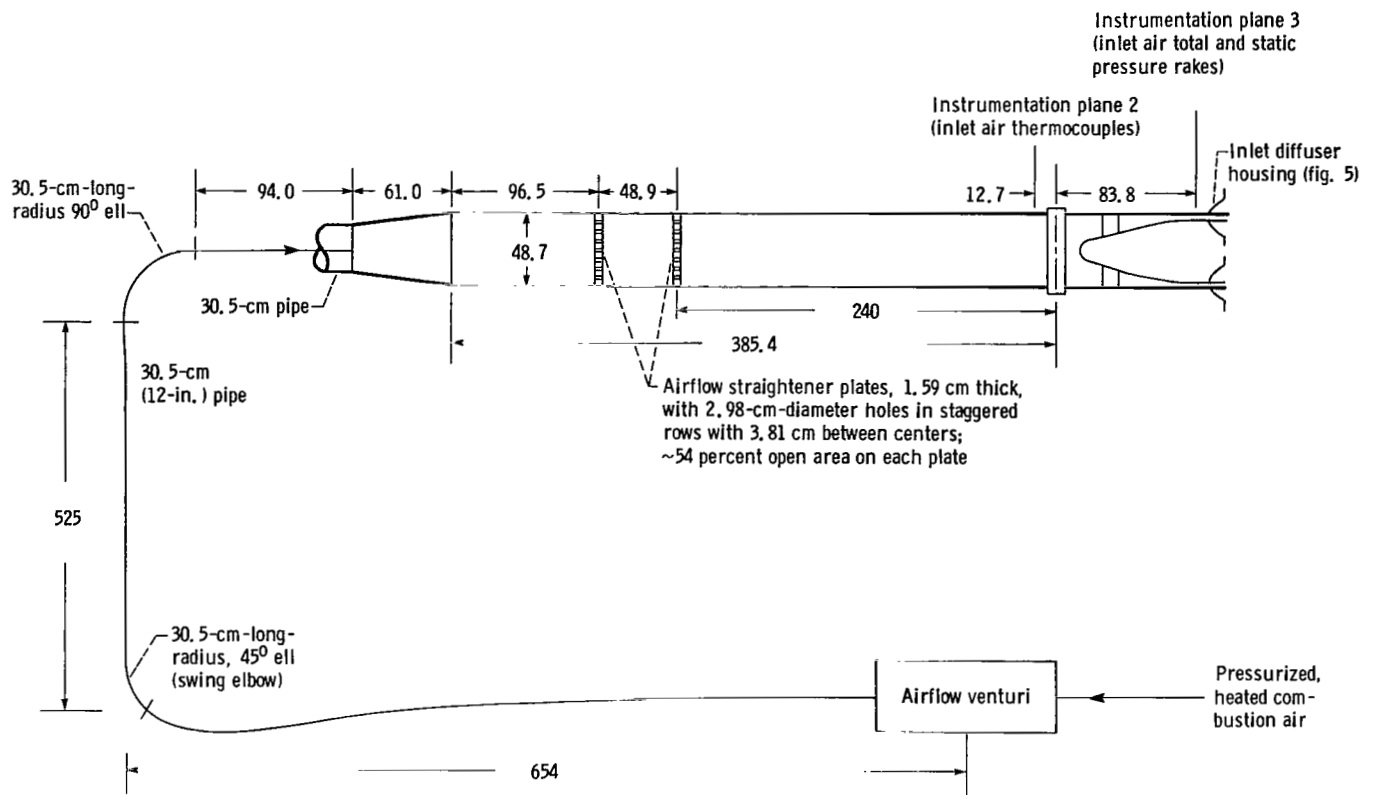


Figure 2. - Inlet piping dimensions and positions of inlet instrumentation planes. (Dimensions are in centimeters. Not to scale.)

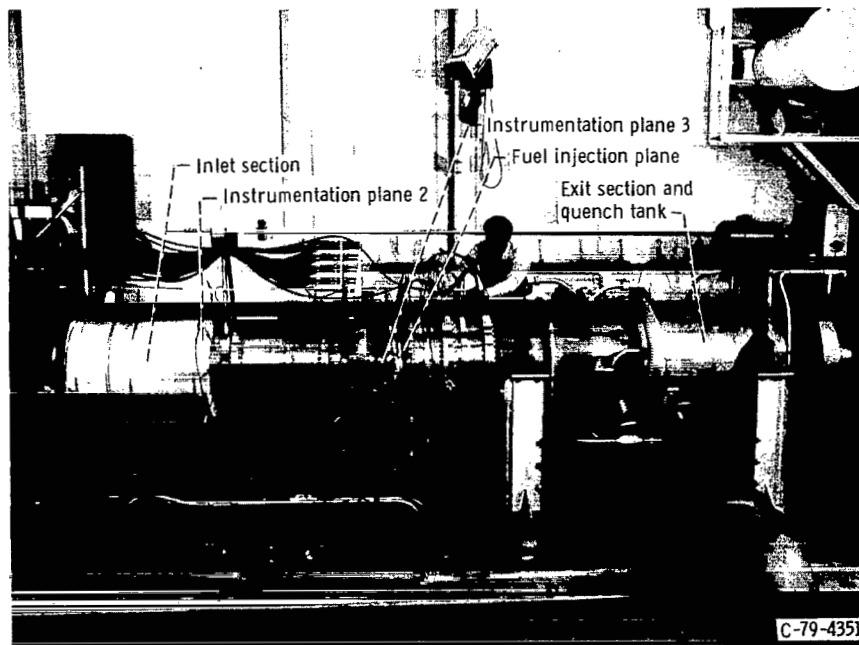


Figure 3. - Combustor test rig.

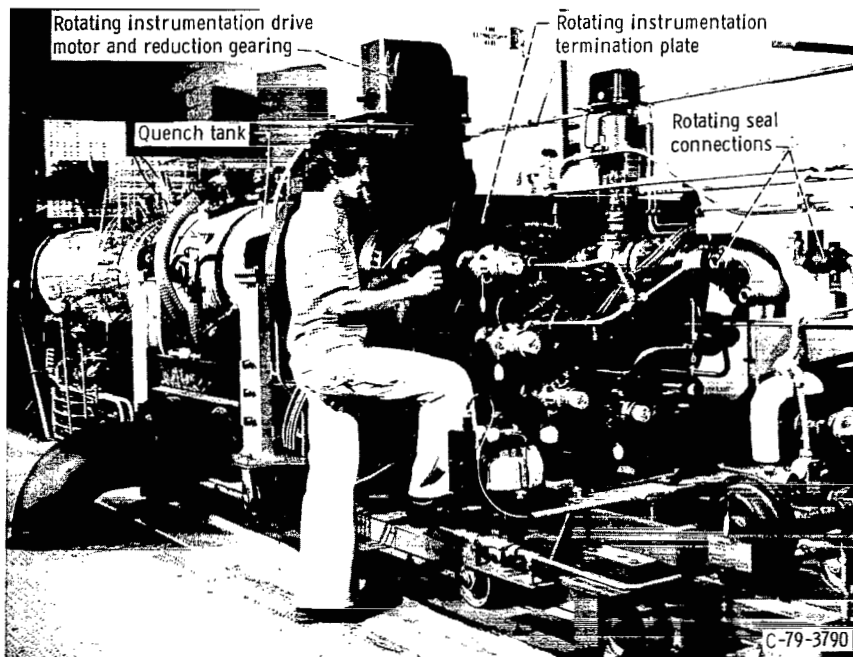


Figure 4. - Combustor rotating instrumentation termination plate and drive motor location.

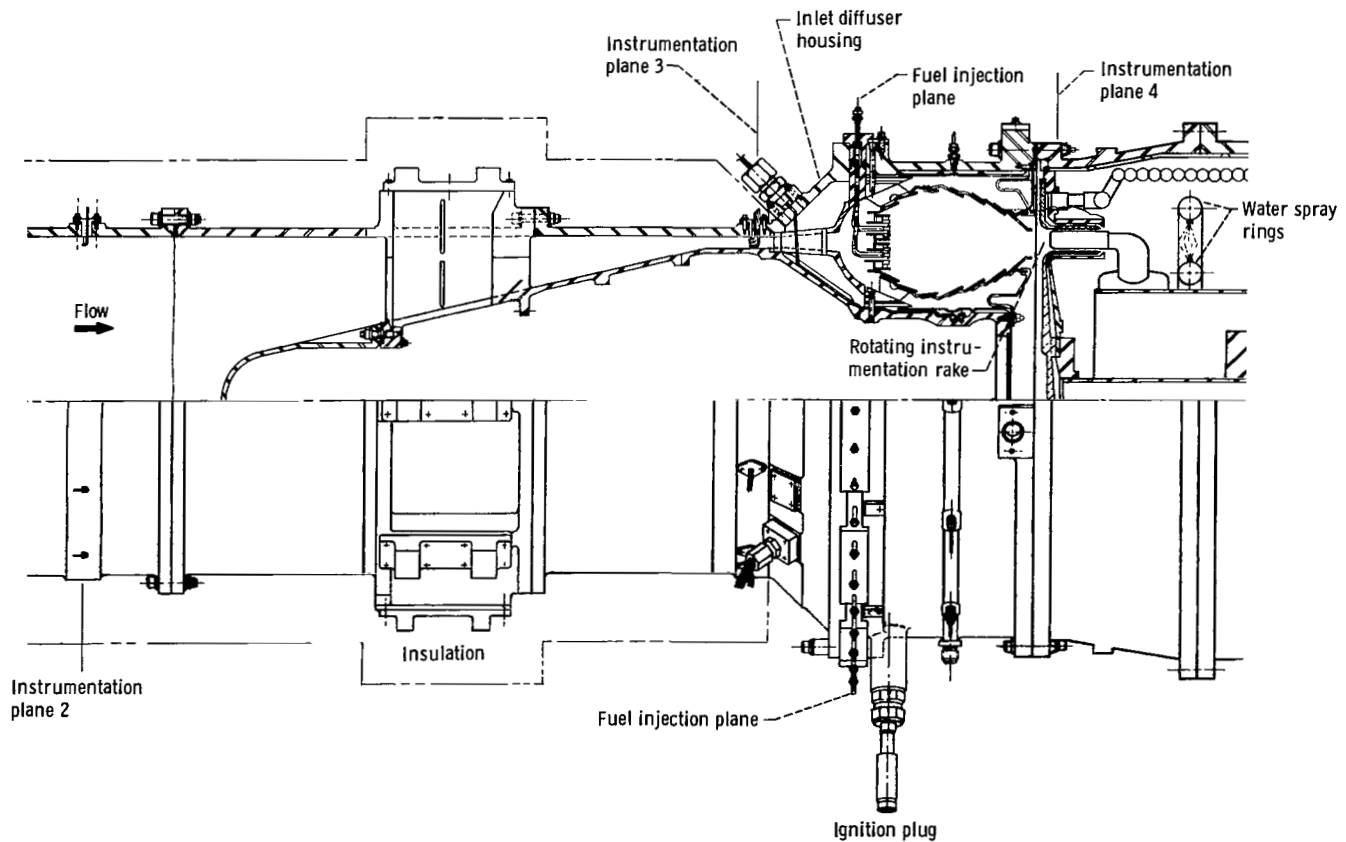


Figure 5. - Test section showing positions of instrumentation planes. (Not to scale.)

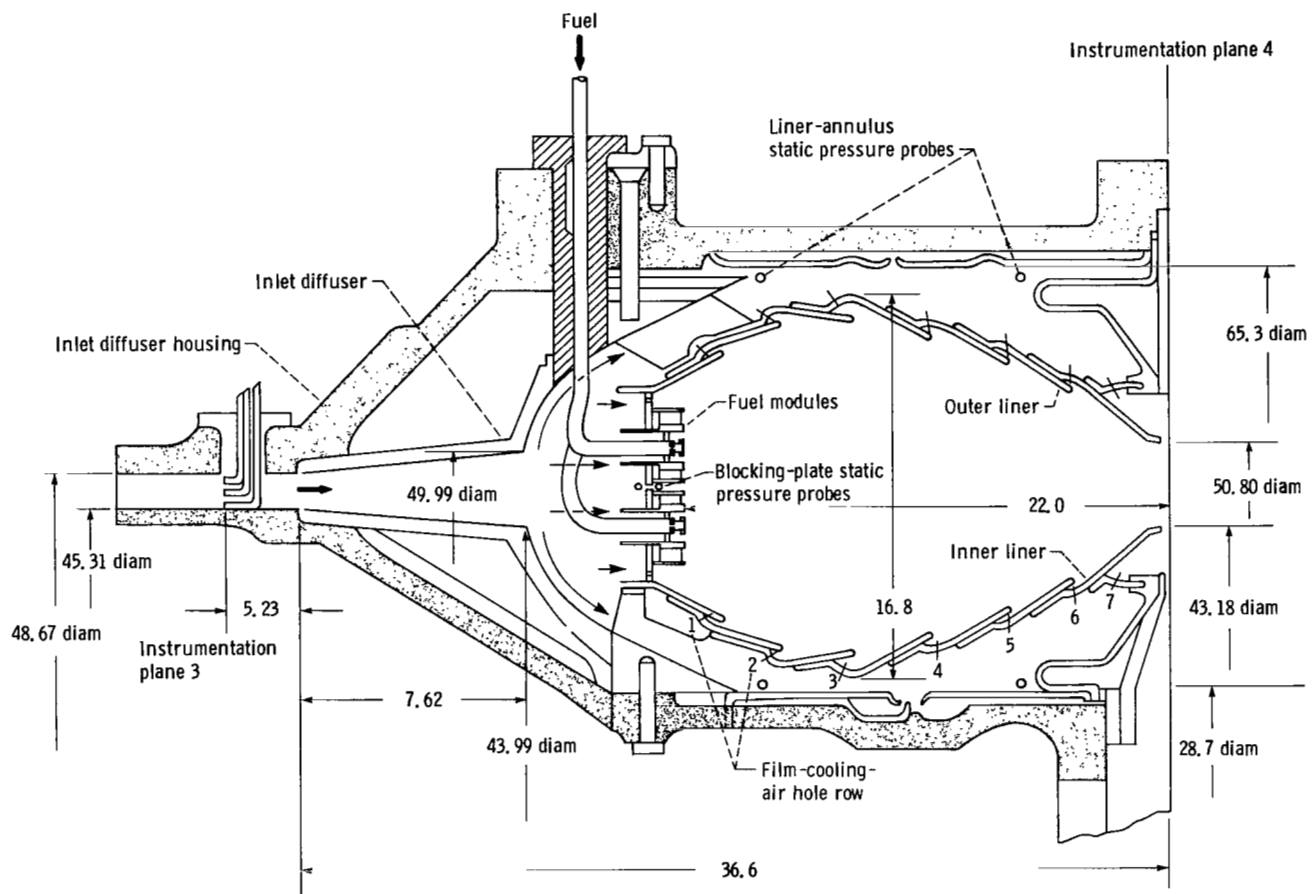
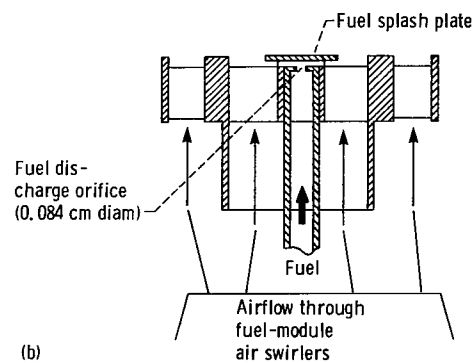
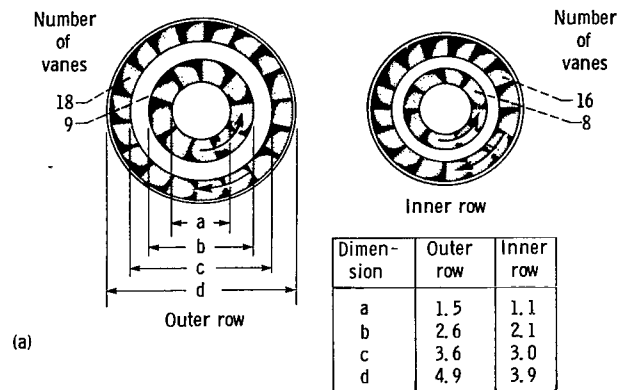
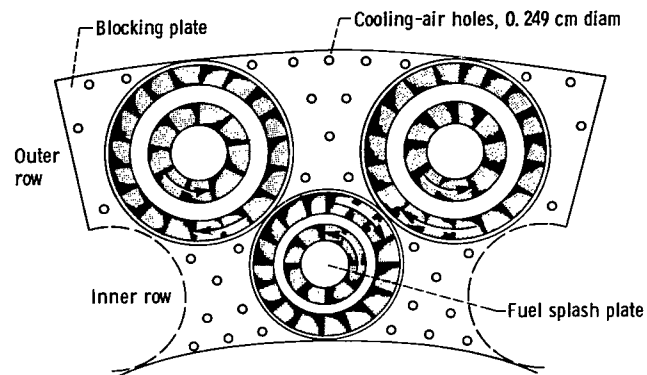


Figure 6. - Cross section of combustion system showing inlet diffuser, inner and outer liners, and fuel modules. (Dimensions are in centimeters. Not to scale.)



(a) View looking upstream, showing air swirlers and air discharge direction.
 (b) View showing airflow and fuel flow passages.

Figure 7. - Fuel modules. (Dimensions are in centimeters.
 Not to scale.)

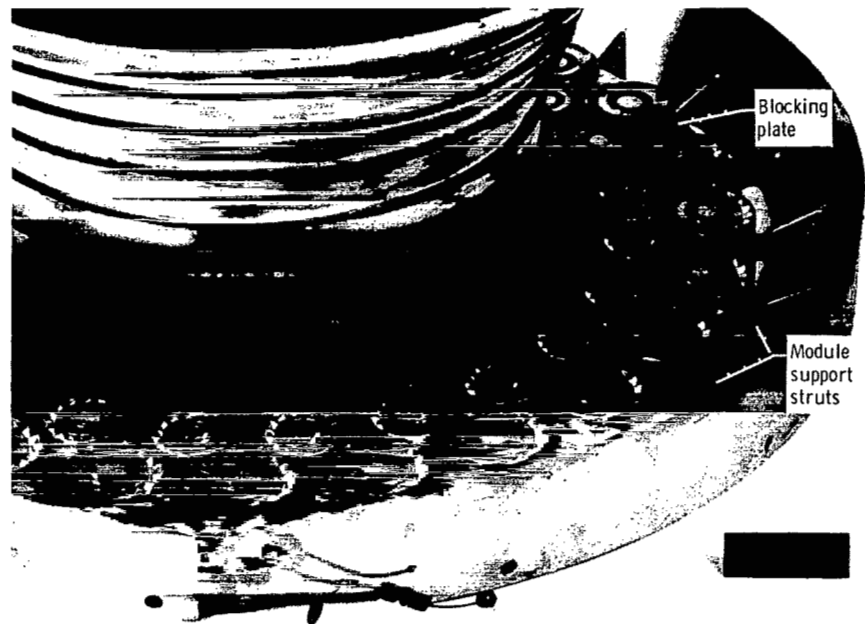


Figure 8. - Two rows of fuel modules, with inner combustor liner installed. (Also shown are fuel-module blocking plate and outer-row module supports.)

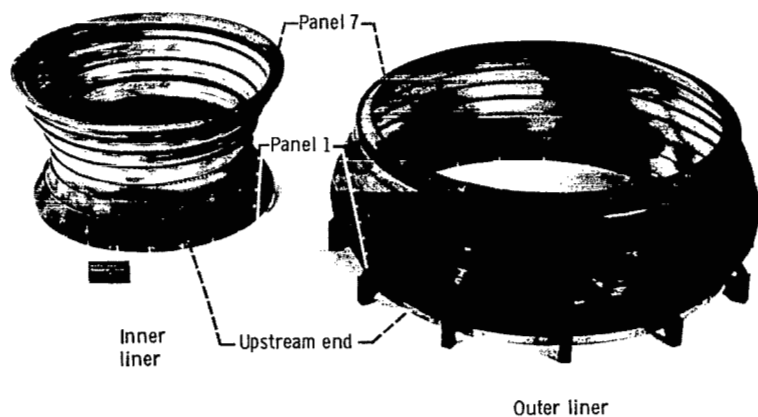


Figure 9. - Combustor liner after completion of tests. (Thermocouple instrumentation has been removed.)

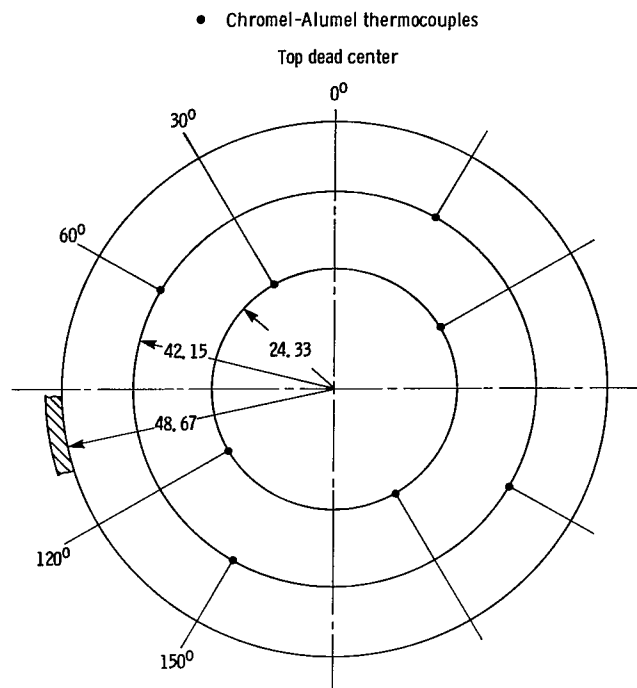


Figure 10. - Temperature instrumentation plane 2 (figs. 2, 3, and 5).
(Diameters are in centimeters. Not to scale. Forward, looking aft.)

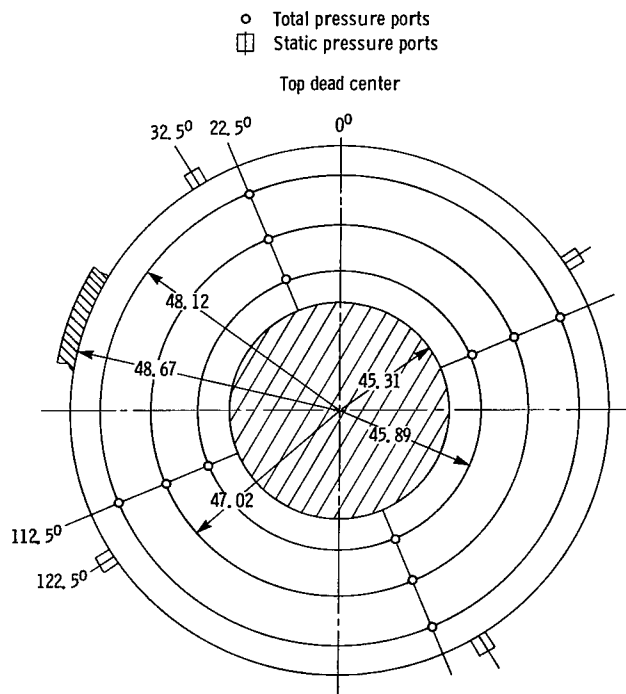


Figure 11. - Pressure instrumentation plane 3 (figs. 2, 3, 5, and 6).
(Diameters are in centimeters. Not to scale. Forward, looking aft.)

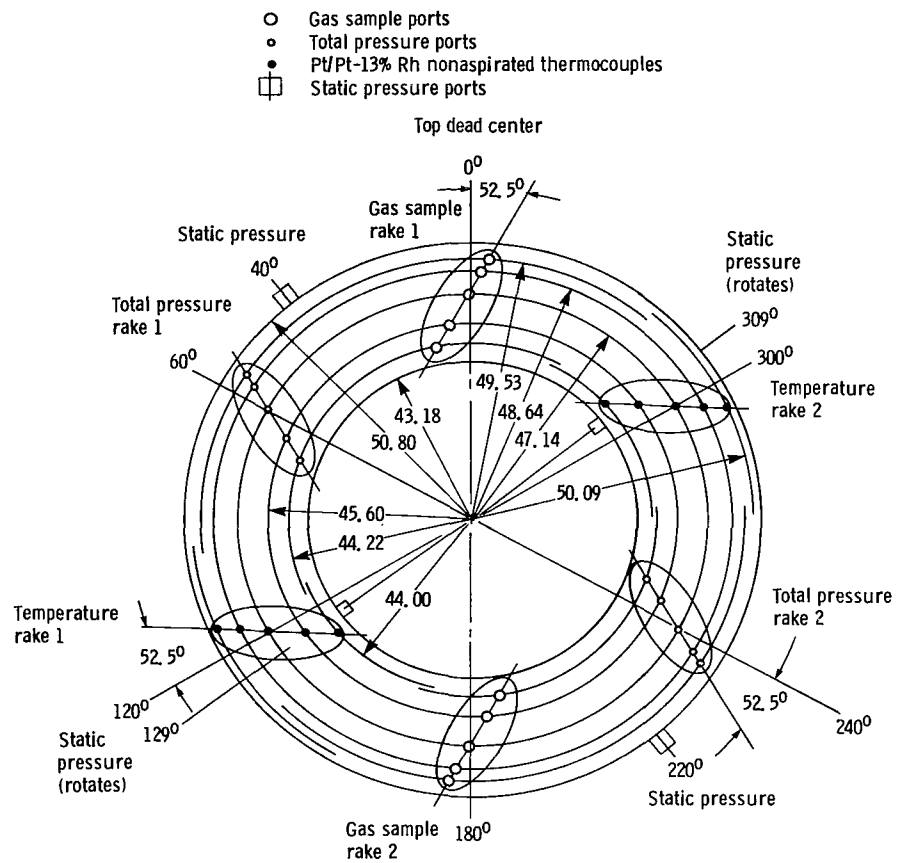


Figure 12. - Pressure, temperature, and gas analysis instrumentation plane 4 (figs. 5 and 6).
 (Diameters are in centimeters. Not to scale. Forward, looking aft.)

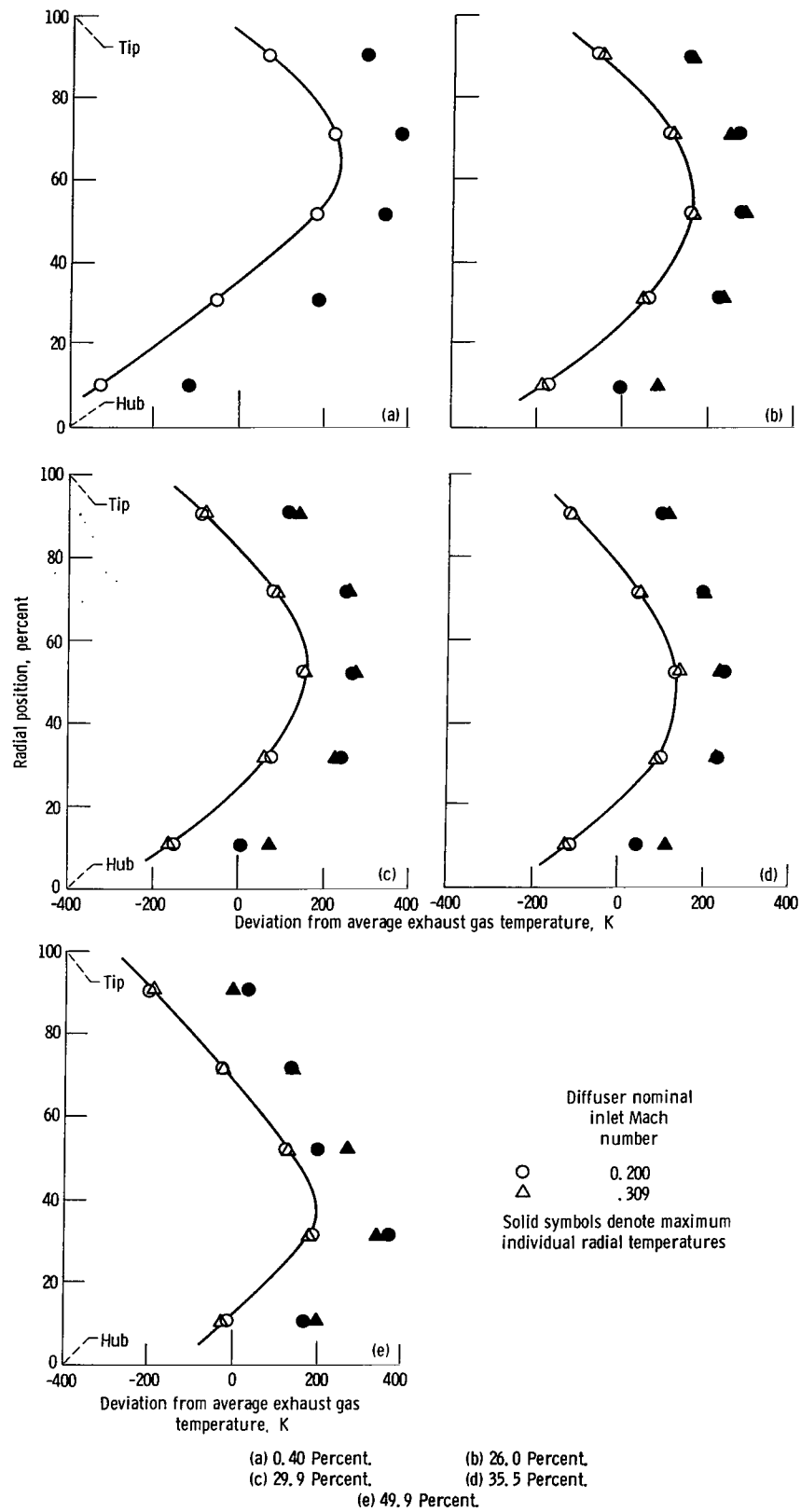


Figure 13. - Radial average and maximum individual temperature profiles at combustor exit for varying percentages of total fuel flow to inner row of fuel modules at following nominal total conditions: inlet pressure, 0.791 MPa; inlet air temperature, 562 K; average exhaust gas temperature, 1172 K.

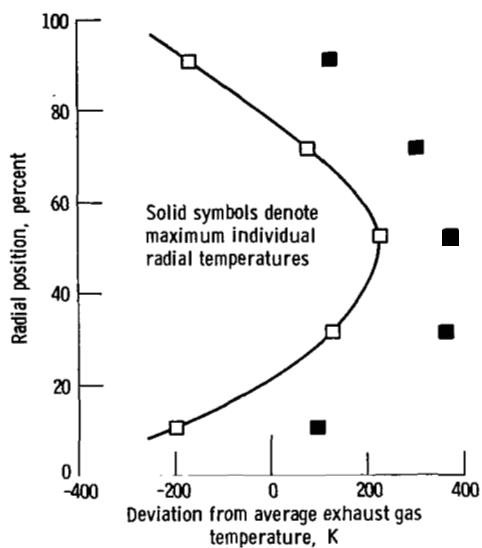


Figure 14. - Radial average and maximum individual temperature profiles at combustor exit at following nominal total conditions: inlet pressure, 0.934 MPa; inlet air temperature, 562 K; average exhaust gas temperature, 1385 K; diffuser inlet Mach number, 0.274; fuel flow to inner row of fuel modules, 33.6 percent of total.

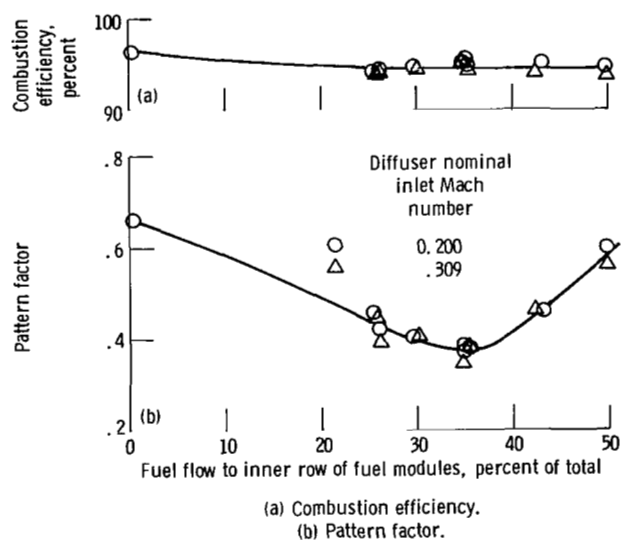
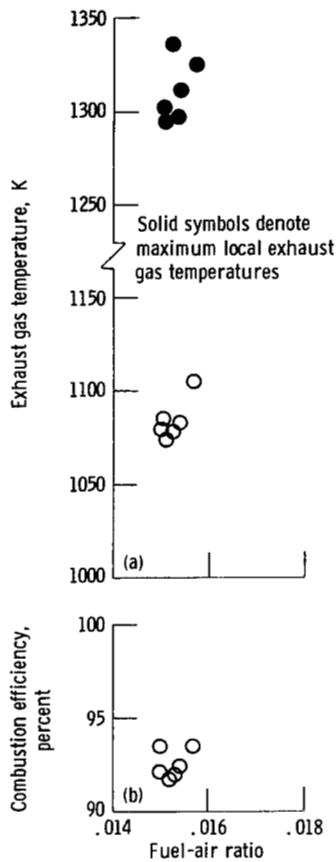
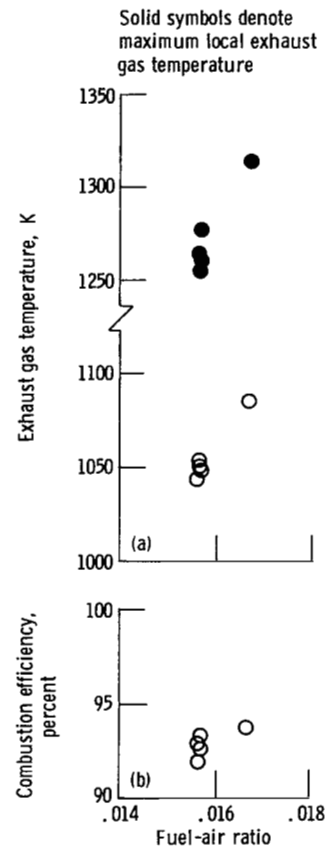


Figure 15. - Effect of fuel flow split between inner row and outer row of fuel modules on combustion efficiency and pattern factor at two diffuser inlet Mach numbers at following nominal total conditions: inlet pressure, 0.791 MPa; inlet air temperature, 562 K; fuel-air ratio, 0.0175.



(a) Average and maximum local exhaust gas temperatures.
(b) Combustion efficiency.

Figure 16. - Effect of fuel-air ratio on average and maximum local exhaust gas temperatures and combustion efficiency at the following nominal total ignition conditions: inlet pressure, 0.792 MPa; inlet air temperature, 556 K; inlet airflow rate, 11.20 kg/sec.



(a) Average and maximum local exhaust gas temperatures.
(b) Combustion efficiency.

Figure 17. - Effect of fuel-air ratio on average and maximum local exhaust gas temperatures and combustion efficiency at the following nominal total idle conditions: inlet pressure, 0.864 MPa; inlet air temperature, 560 K; inlet airflow rate, 17.98 kg/sec.

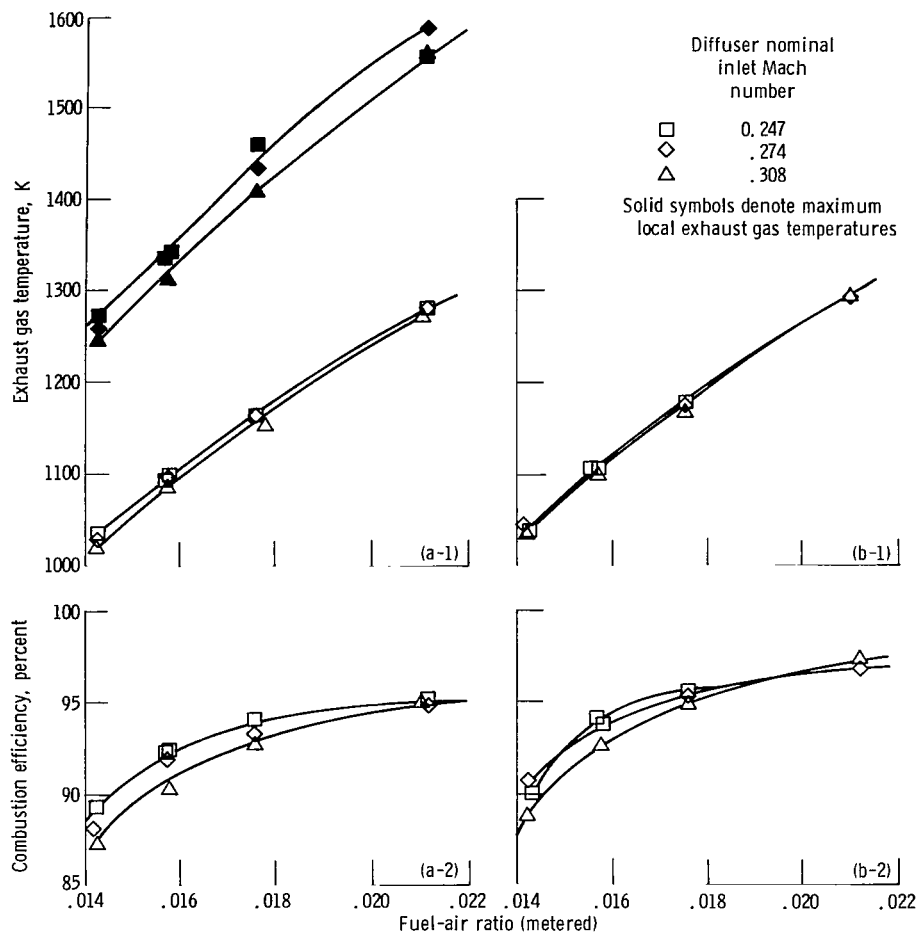
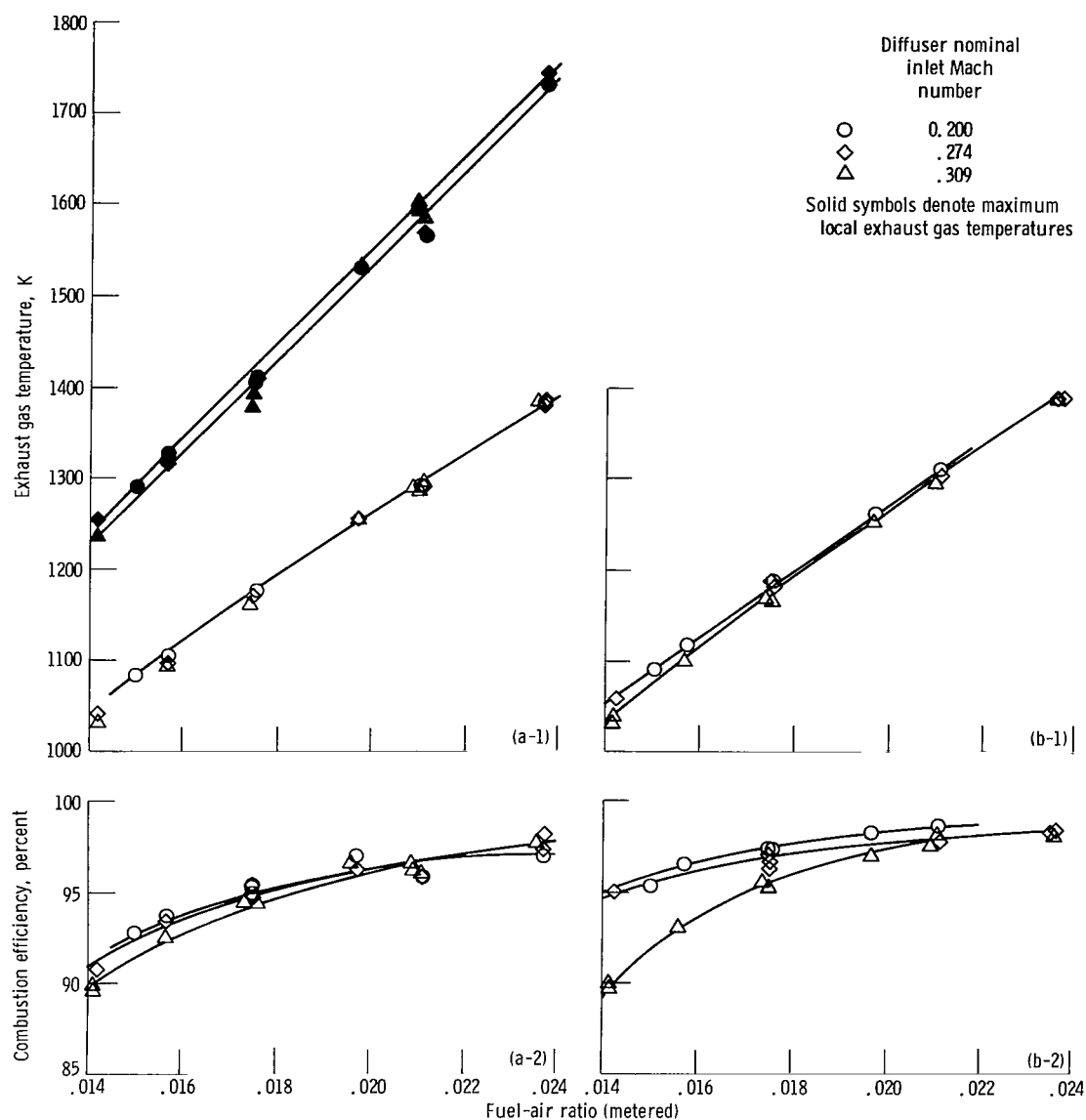


Figure 18. - Effect of fuel-air ratio and diffuser inlet Mach number on average and maximum local exhaust gas temperatures and combustion efficiency at the following nominal total conditions: inlet pressure, 0.648 MPa; inlet air temperature, 562 K.



(a-1) Average and maximum local exhaust gas temperatures.
 (a-2) Combustion efficiency.
 (a) Exhaust gas thermocouples used to obtain performance data.

(b-1) Average exhaust gas temperature.
 (b-2) Combustion efficiency.
 (b) Gas analysis used to obtain performance data.

Figure 19. - Effect of fuel-air ratio and diffuser inlet Mach number on average and maximum local exhaust gas temperatures and combustion efficiency at the following nominal total conditions: inlet pressure, 0.792 MPa; inlet air temperature, 563 K.

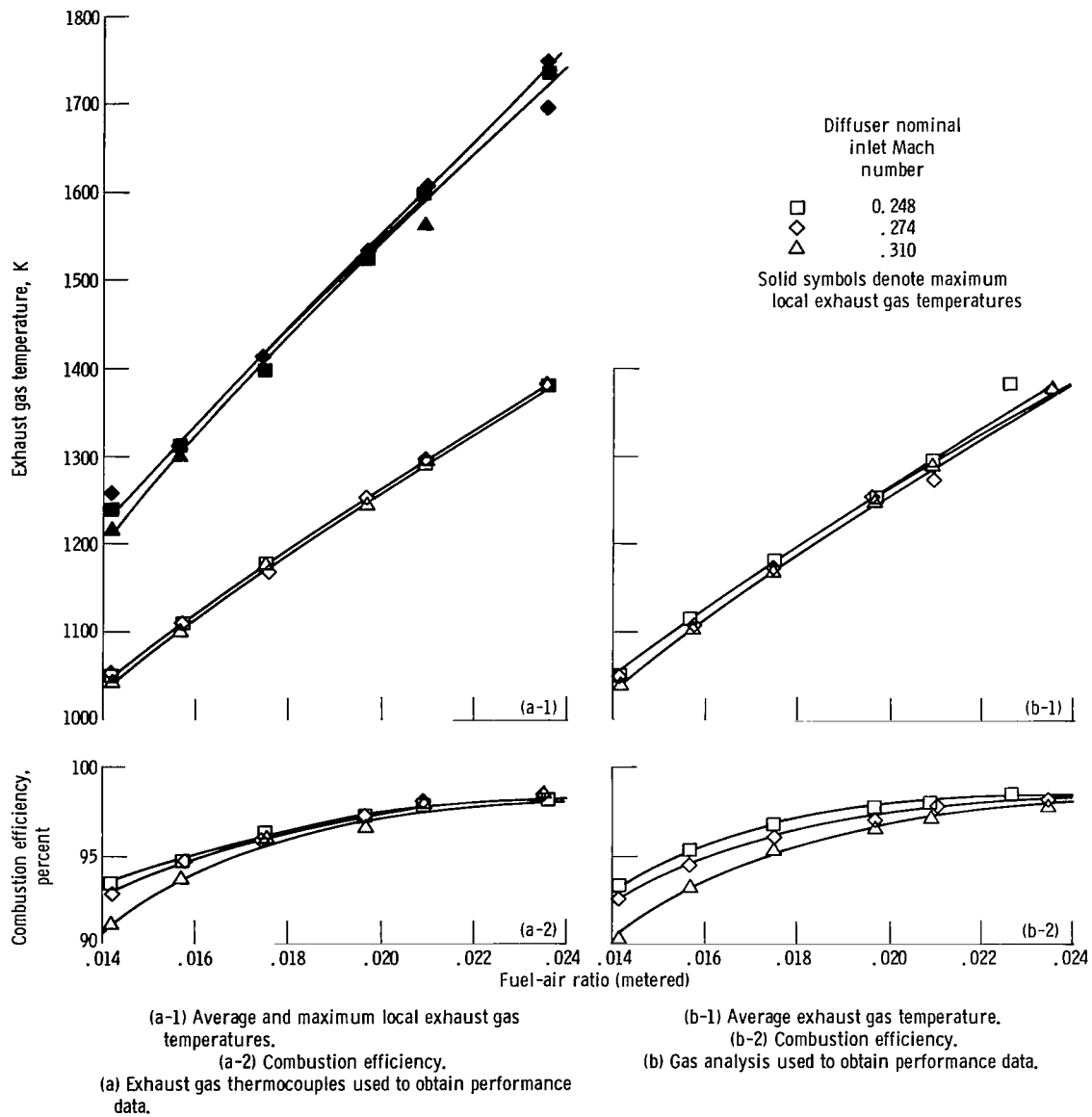
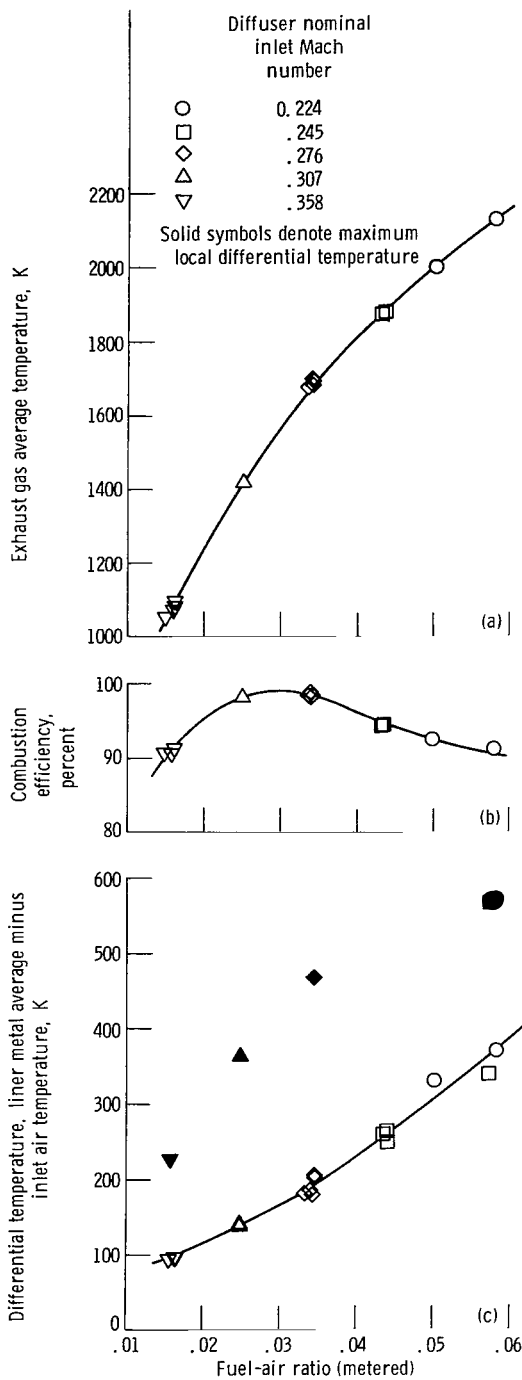
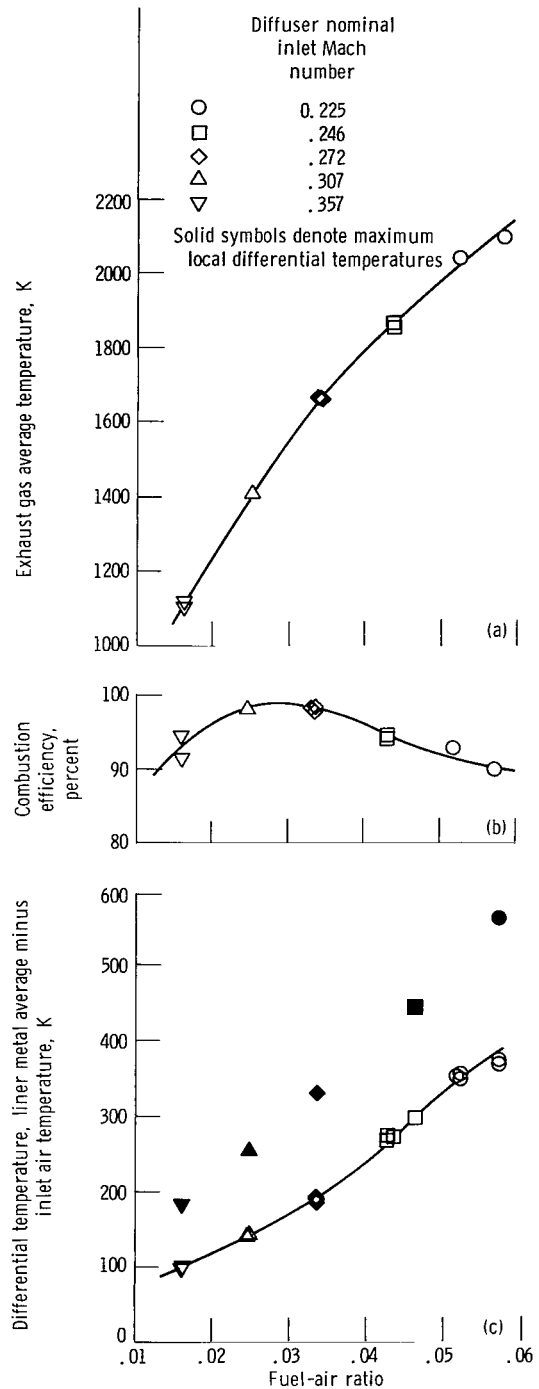


Figure 20. - Effect of fuel-air ratio and diffuser inlet Mach number on average and maximum local exhaust gas temperatures and combustion efficiency at the following nominal total conditions: inlet pressure, 0.933 MPa; inlet air temperature, 563 K.



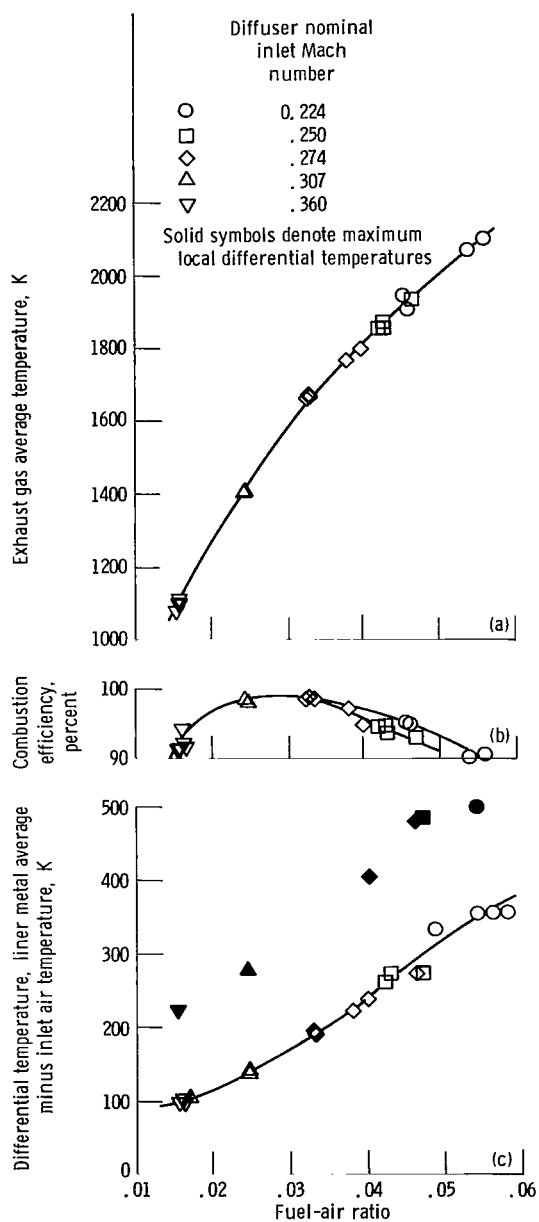
(a) Average exhaust gas temperature.
(b) Combustion efficiency.
(c) Combustor liner average and maximum differential temperatures.

Figure 21. - Effect of fuel-air ratio and diffuser inlet Mach number on average exhaust gas temperature, combustion efficiency, and liner average and maximum differential temperatures at the following nominal total conditions: inlet pressure, 0.650 MPa; inlet air temperature, 562 K; turbine inlet Mach number, 0.245.



(a) Average exhaust gas temperature.
(b) Combustion efficiency.
(c) Combustor liner average and maximum differential temperatures.

Figure 22. - Effect of fuel-air ratio and diffuser inlet Mach number on average exhaust gas temperature, combustion efficiency, and liner average and maximum differential temperatures at the following nominal total conditions: inlet pressure, 0.795 MPa; inlet air temperature, 563 K; turbine inlet Mach number, 0.230.



(a) Average exhaust gas temperature.
(b) Combustion efficiency.
(c) Combustor liner average and maximum differential temperatures.

Figure 23. - Effect of fuel-air ratio and diffuser inlet Mach number on average exhaust gas temperature, combustion efficiency, and liner average and maximum differential temperatures at the following nominal total conditions: inlet pressure, 0.938 MPa; inlet air temperature, 562 K; turbine inlet Mach number, 0.232.

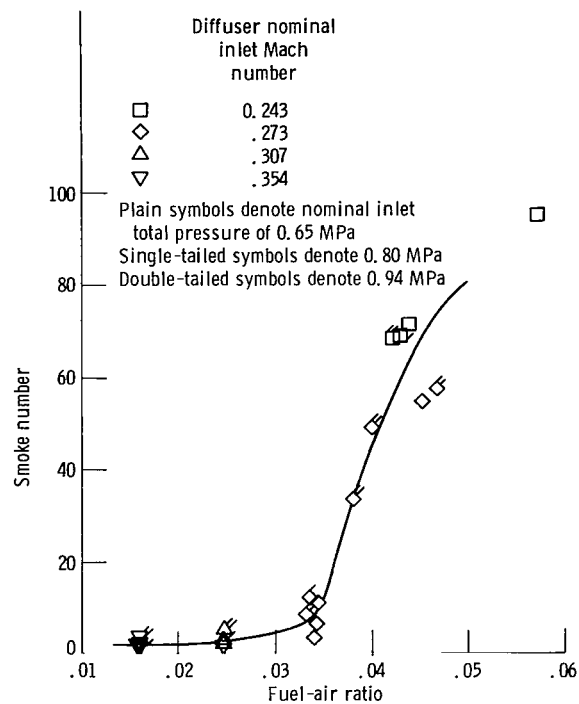


Figure 24. - Comparison of smoke numbers obtained over a range of fuel-air ratios at different inlet pressures and diffuser inlet Mach numbers at the following nominal total conditions: inlet air temperature, 562 K; turbine inlet Mach number, 0.237.

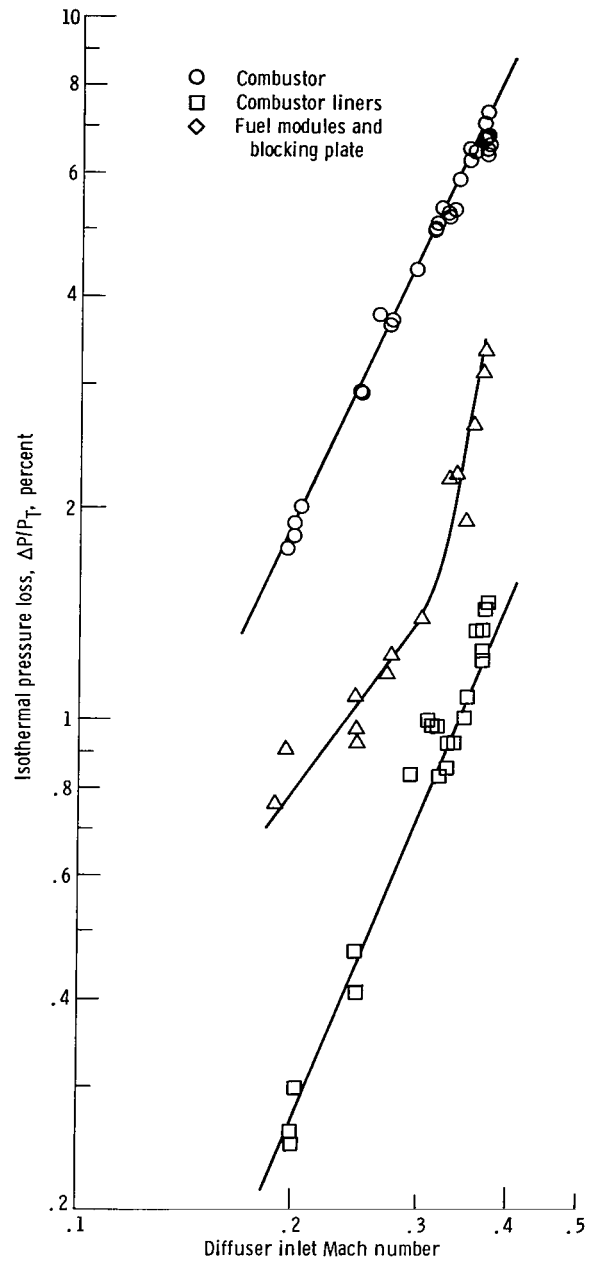


Figure 25. - Effect of diffuser inlet Mach number on isothermal (nonburning) pressure loss across entire combustion system, fuel modules and blocking plate, and combustor liner. Variable inlet air pressure, temperature, and flow rate.

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16. Abstract A combustion system has been developed to operate efficiently and with good durability at inlet pressures to 4.05 MPa (40 atm), inlet air temperatures to 900 K, and exhaust gas temperatures to 2480 K. A preliminary investigation of this system was conducted at inlet pressures to 0.94 MPa (9 atm), a nominal inlet air temperature of 560 K, and exhaust gas temperatures to 2135 K. A maximum combustion efficiency of 98.5 percent was attained at a fuel-air ratio of 0.033; the combustion efficiency decreased to about 90 percent as the fuel-air ratio was increased to 0.058. An average liner metal temperature of 915 K, 355 kelvins greater than the nominal inlet air temperature, was reached with an average exhaust gas temperature of 2090 K. The maximum local metal temperature at this condition was about 565 kelvins above the nominal inlet air temperature and decreased to 505 kelvins above with increasing combustor pressure. Tests to determine the isothermal total pressure loss of the combustor showed a liner loss of 1.1 percent and a system loss of 6.5 percent.		13. Type of Report and Period Covered Technical Paper
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